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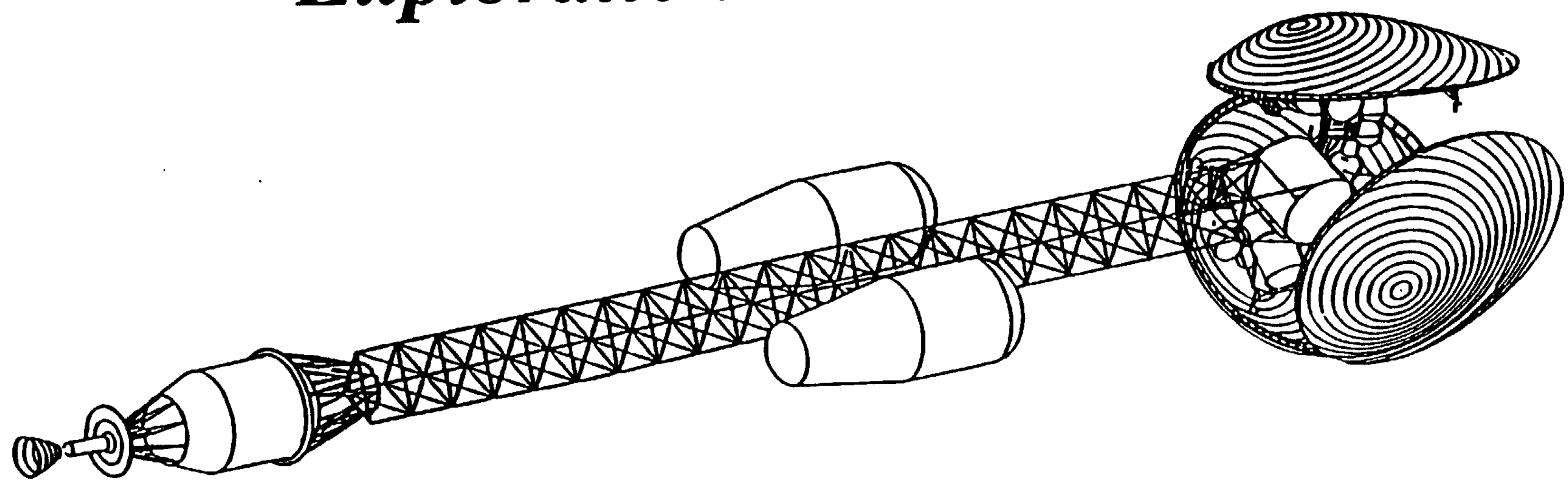
Space Transfer Concepts and Analysis for Exploration Missions



Implementation Plan and Element Description Document

Volume 1: Major Trades Book 2 (draft Final)

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ADVANCED CIVIL SPACE SYSTEMS

Boeing Aerospace and Electronics
Huntsville, Alabama

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Artificial Gravity Assessment Study

This section assesses the STCAEM reference vehicles' (CAB, NTR, SEP and NEP) adaptability to artificial gravity (g_a). Penalties for each case are presented with a final mass comparison. Human factors assumptions for the study are based on historical studies on human adaptability to artificial gravity.

The CAB configuration employs a planar beam armature with three conductive, ribbon-section tethers. Communications and power are located, despun, at the CM for constant tracking capabilities. The CAB configuration has approximately a 15 % mass penalty over the μg version because of the added components and propellant required. Packaging the undeployed reel/crawler mechanisms requires a slightly larger aerobrake than for the μg case. If the MTV aerobrake is not retained (that is, for non-reusable mission scenarios), a very long tether system (> 2 km) is required on the return trip.

The NTR configuration is the least affected by artificial gravity requirements. The main change in the artificial gravity configuration is a lengthened truss to allow a 56 m radius to the transfer hab. The drop tanks are positioned at the CM so that the center of rotation does not move as they are dropped; no deployable truss or tether is needed. The mass penalty for the NTR artificial gravity vehicle is on the order of less than or equal 10 %.

Artificial gravity for continuous thrust systems (SEP and NEP) is not as simple as for cryo and nuclear thermal systems because of the "spinning-while-thrusting" problem. Either high power spin-joints, or cross-product engine assemblies tend to be required. A possible solution to this problem is to fly in μg for most of the trip, spinning up only at mid-course no-thrust intervals and upon arrival at Mars. On conjunction class missions, where stay times at Mars are up to 600 days, the vehicle can be spun-up in Mars orbit to recondition the crew prior to landing. 7 SEP and 2 NEP options have been evaluated, and a new concept has been selected, called an eccentric rotator. This avoids virtually all g_a penalties for SEP, but still requires high-power roll rings for NEP. Preliminary estimates of mass penalties for EP vehicles are of order 5 %.

PT-2

Artificial Gravity (g_a) Assessment

Assumptions	Rationale
1g gravity level	• Earth-normal conditioning for exploration in surface EMU
Rotation rate ≤ 4 rpm (56 m)	• Generally accepted range for vestibular disturbance tolerance
Contiguous crew compartments	• Maximize available volume • In-flight simulation and training • Contingency operations
Truss and tether connections • Tethers are "ribbon" shaped	• Avoids mass penalty • Not needed for contiguous volumes • Facilitates conductors
Module orientation parallel to spin vector	• g level consistency; minimizing vestibular disturbance • Mass properties quasi-isotropic to first order

Cryogenic/Aerobrake Vehicle Artificial Gravity Configuration

The CAB artificial gravity configuration uses tethers to achieve the 56 m spin radius required to produce 1g at 4 rpm. This rotation rate is currently the maximum thought allowable to avoid transient vestibular disturbances for most people. The tethers used are conductive tethers to simplify the mechanics of power transmission from a despun photovoltaic array to the end-mass vehicles. The tether is "ribbon" shaped to reduce the possibility of entanglement during the reeling cycles, to better facilitate "crawler" operations, and because it radiates heat (generated resistively during power transmission) better than a circular cross-section of the same sectional area.

The configuration is a planar "beam" arrangement of three tethers, with crawlers and despun solar array and communications laser located at the CM. The MTV propulsion system is split symmetrically, forming a yoke around the habitat system. This allows the habitat system to be detached and removed simply from it original position. Post-TMI, the transfer hab separates in this manner, remaining contiguously connected with the MEV; the MTV aerobrake and TEI propellant become the g_a countermass. The Mars to Earth configuration uses the MTV aerobrake and the *empty* TEI propellant tanks as countermass, which necessitates a longer countermass radius to maintain a 56 m separation between the crew systems and the center of rotation (CM). In a nonreusable scenario, the MTV aerobrake is jettisoned at Mars; this saves TEI propellant but requires a much longer countermass tether radius for the return trip: over 2 km.

The crawler/mast/power assembly at the CM includes deployable trusses that separate the tethers into the wide-beam spin-configuration, yet package tightly for stowage beneath the MTV habitat system for μg mission phases (including aerobraking). The solar array and the communications laser are on despun joints for independent tracking of the sun and Earth. The crawler mechanism is divided into two sections, so that one section can always be at the CM to support the deployable truss and the tether while the other is performing its transportation function. Each crawler has small solar arrays for its motive power. When stopped at the CM, each crawler contacts exposed portions of the aluminum conductors inside the tethers, to transfer power from the solar array to the end-mass vehicles.

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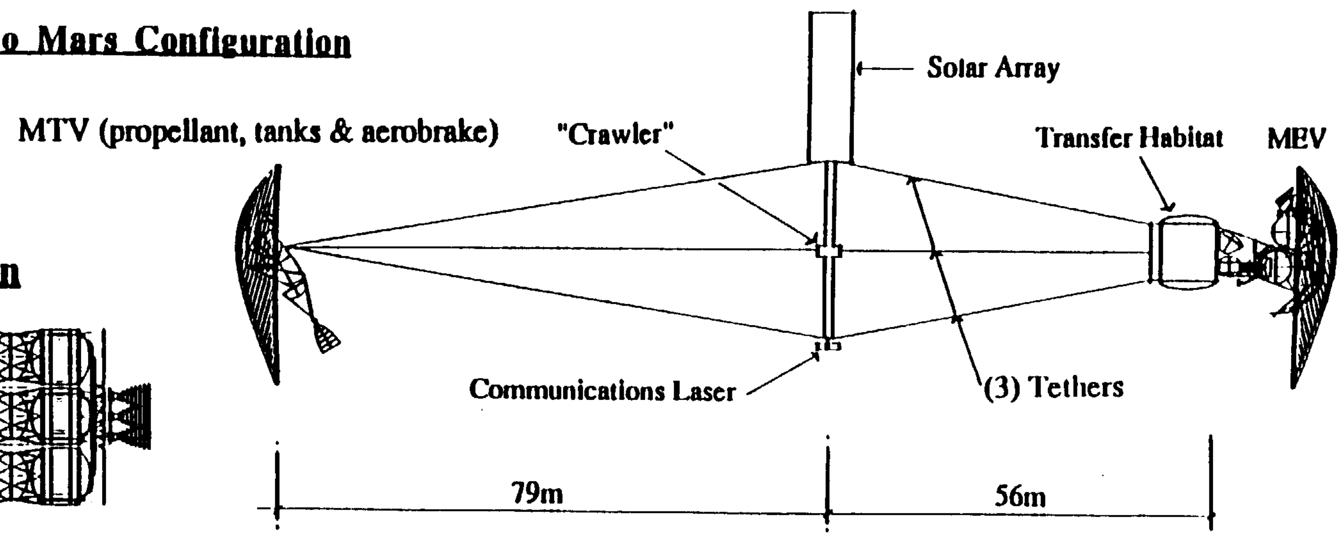
The g_a CAB mass penalty, when compared to a reusable μg version, is ~ 15 %, because of the hardware and spin-up/down propellant required to support artificial gravity operations. Accommodating the tether reel, crawler, solar arrays, and communications laser below the transfer habitat requires a 32 m aerobrake, slightly larger than the 30 m baseline CAB brake. Technology penalties include accessible-conductor tethers, and rotating joints for the solar array and communications laser. Operations penalties include maintenance of the mechanical systems required for g_a, and the EVA complications associated with using tether crawlers for end-to-end mobility.

g_a Cryo/AB Configuration

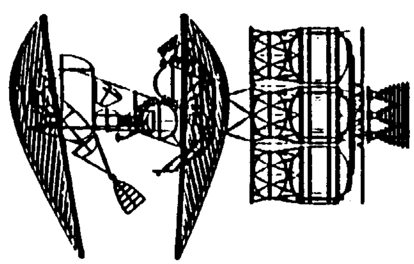
This chart shows the 3 main configurations of the vehicle in transit. The Earth to Mars phase requires a total tether length of 128m, while the Mars to Earth phase requires a total tether length of 161m with MTV aerobrake, and 2.15km without MTV aerobrake. The solar array and the communications laser are located at the CM on a "despun" joint to track the sun and Earth respectively. The crawler is also located at the CM nominally, but has the ability to travel to either end of the central tether to transfer crew and/or supplies. The initial TMI configuration is shown for comparison.

g_a Cryo/AB Configuration

Earth to Mars Configuration

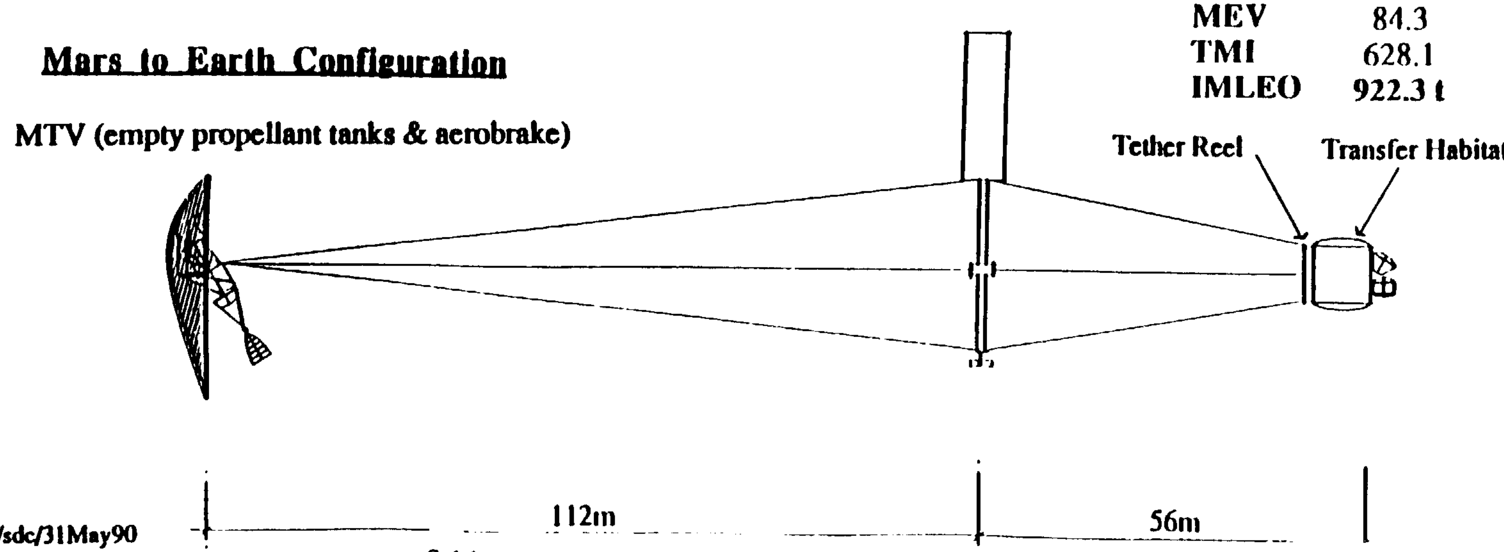


TMI Configuration



	ECCV Return	Earth Aerocapture
MTV	209.9	239.2
MEV	84.3	84.3
TMI	628.1	690.6
IMLEO	922.3 t	1,014.1 t

Mars to Earth Configuration



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Art-g (1-g) Chem/aerobrake Vehicle for the 2015 Opposition Mission

4 RPM tether sys, 4 spinup/down maneuvers, Crew of 4, ECCV return, 565 day total trip time

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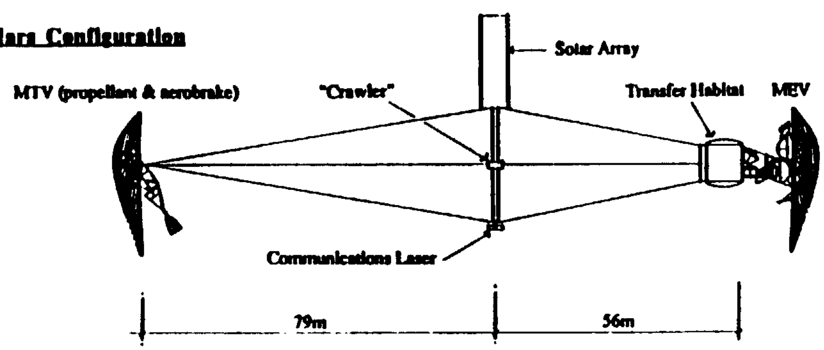
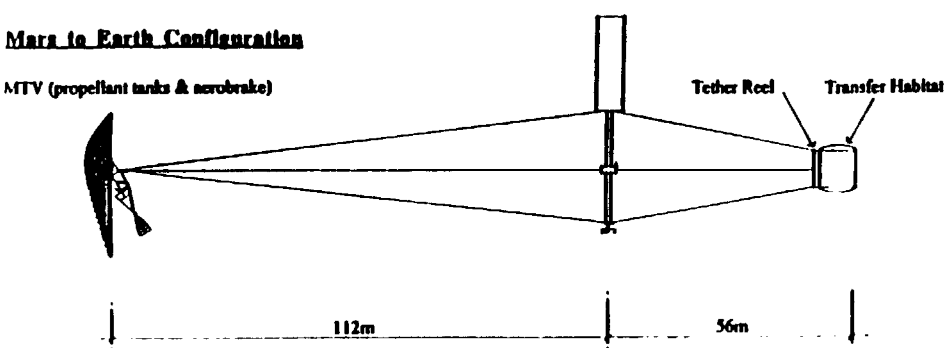
Earth to Mars Configuration



Mars to Earth Configuration



Element	*ECCV return*	*Earth Aerocapture*
MTV Mars aerobrake	23758	23758
MTV crew hab module 'dry'	28531	28531
MTV consumables & resupply	7096	7096
MTV science	1000	1000
MTV propulsion stage	21847	27473
MTV MPS propellant load	103129	132177
*MTV Art-g added hardware wt	8057	8057
*MTV Art-g added RCS prop	9004	10627
MTV total	**202422**	**238719**
MEV Mars capture & desc aerobrake	15138	15138
MEV ascent stage	22754	22754
MEV descent stage	21457	21457
MEV surface cargo	25000	25000
MEV total	**84349**	**84349**
ECCV	**7000**	**0**
Cargo to Mars orbit only	0	0
MTV-TMI interstage wt	500	500
TMI inert stage wt	62820	69080
TMI propellant load	565270	621540
TMI stage total	**628090**	**690620**
IMLEO	**922,361**	**1,014,188**

** Artifical -g system weight penalties* *all masses in kg*
Earth aerocapture configuration shown in diagram

Mac chart: M 1-g chem/ab cover pg
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Mars dep stg - for Artificial-g (1-g) 2015 Chem/Aerob Veh

4 RPM tether system, 4 spiuup/down maneuvers, Crew of 4, 2 adv eng's; Isp = 475

6/5/90

		Element	mass (kg)	Rationale
TEI stage inert	[154]	Fuel tank	6545	2 SiC/Al metal matrix tanks, 37 ksi working stress, tank MEOP=175kPa
	[155]	Oxygen tank	3769	2 SiC/Al metal matrix tanks, 37 ksi working stress, tank MEOP=175kPa
	[158]	MLI/meteor shield	1282	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm. Meteor Shield: 2 (kg/m2)
	[159]	Frame structure	6243	5% of MTV propellant + 5% of MTV stg inert mass
	[518]	Main propulsion	794	2 x 30k lbf advanced space eng's: Isp=475 s, high AR nozzle not extendible
	[183]	Maneuver RCS inert	300	Scaled from RCS propellant
	[160]	<u>Mass growth</u>	<u>2914</u>	15% growth for inert stage
	[161]	***Mars dep stg 'dry' wt***	21847	
Art-g spin equip		Added wt necessitatedby Art-g		
		• tethers	2178	3 at 180 m each
		• tether reel	3050	Attached to MTV hab
		• tether crawler	1500	Transverses tether length, centers solar arr at veh/tether sys Cg for despun operation
		• lock joints & other equip	829	Lock joint secures MTV mod to MTV propulsion stg during MPS thrusting
		• <u>added Spin RCS sys</u>	<u>500</u>	Spinup/down RCS thrusters, lines, tanks, etc above the nominal maneuver RCS
		Total Art-g only hardware	8057	Total Art-g hardware penalty not required for a zero-g vehicle
Art-g spin prop		Art-g spin RCS propellant		1-g Art-g; See diagram of inflight spinup phases.
	[224/225]	• outbound 1st spinup/down	3298	Each of 2 counter wts (hab mod+MEV & MTV 'wet' propul stg+AB) spun to 4 RPM
	[222/223]	• outb 2nd spinup/down	3258	Despun for outbound midcourse corection MPS burn, then respun to 4 RPM
	[220/221]	• inb 1st spinup/down	1224	MEV left behind, MTV propul stg 'dry' except for inb midc correction burn
	[220/221]	• <u>inb 2nd spinup/down</u>	<u>1224</u>	
	Sum	**Total Art-g only RCS prop**	9004	Total Art-g RCS propellant penalty
TEI prop & boil-off	[118]	RCS maneuver propellant	599	Gaseous O2/H2 propellant, Isp=400 sec, MTV maneuver RCS dV=30 m/sec
	[122]	MTV inb midcourse burn prop	1516	delta V: 90 (m/sec); burn done with MTV Mars dep main propulsion
	[128]	Mars dep usable prop	87732	LH2/LO2, MR=6:1, *Mars dep dV: 3400 m/sec* usable=prop req after outb & inorbit
	[545+546]	In orbit Mars dep prop boiloff	485	boiloff; 30 day boiloff period; calculated with Boeing's 'CRYSTORE' program
	[121]	Outb midcourse burn prop	7638	midcourse maneuver delta V: 120 (m/sec); burn done w MTV main propulsion
	[498+499]	<u>Outb Mars dep prop boiloff</u>	<u>5159</u>	335 day outbound trip time.
	Sum	***tot MTV propellant load***	103129	
	[556]	***Tot M dep propulsive stg wt (at time of E dep burn)***	142037	

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Mac chart M 1-g MTV veh wt-rationale

TMI stg - MTV for Artificial-g (1-g) 2015 Chem/Aerobrake Veh

4 RPM, 4 spinup/down maneuvers, ECCV Return, 4 x 200k lbf adv eng's: Isp = 475

6/5/90

		Element	mass (kg)	Rationale
TMI Pay load	[556]	Tot MTV Mars dep stg	142037	See mars dep stage wt statement
	[380+179]	MTV Crew hab mod sys	36627	See MTV crew hab module wt statement
	[230]	ECCV	7000	4 man apollo type entry vehicle; MTV expended
	[106]	MEV	84349	4 man, 30 day stay, 25 t surface cargo
	[159]	Outb 'to-Mars-orbit' cargo	0	communication sat's taken on precurser mission
	[1292]	Mars Site Recon Vehicle	0	Not taken for Ref 2015 mission
	[163]	MTV-TMI interstage wt	500	Structural member joining TMI to MTV
		MTV Mars capture aerobrake:		Structural design assumptions: 200ksi spar strength, 22.5 inch spar depth
		• Primary spar weight	4239	
		• Secondary spar wt	3434	
		• Honeycomb wt	12785	
		• TPS wt	3300	
	[169]	Total:	23758	
	[168]	***Tot TMI stg 'Payload wt'***	294271	TMI stage injects this 'payload' wt into Mars hyperbolic trajectory
TMI stage	[172-173]	TMI stage inert	62820	0.9 propellant fraction
	[173]	TMI propellant load	565270	TMI stage tanks topped off before ignition, no boiloff accounted for
	[172]	***TMI stage total mass***	628090	4 x 200k lbf advanced spaace engines, Isp=475 sec
	[171]	***IMLEO initial mass in low Earth orbit***	922361	

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Mac chart M 1-g TMI wt-rationale

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Conductive Tethers

Conductive tethers have been used in this study to simplify the power transmission method. Conductive tethers are not a simple technology as demonstrated by the examples given on the following chart. Conductive tethers also simplify the reeling process because of the reduce number of cables.

Conductive Tethers

- Transmitting electric power through long flexible cables is standard practice on Earth
  - technology issues well understood
- Maintaining electrical contact between the mobile "crawler" and the tether
  - similar to track lighting, sliprings, electric motors and generators, electric subway trains, and trolleys
  - a Solar Power Satellite concept (1980) incorporated a 5 GWe slipring
  - partially despun spacecraft use lower power sliprings regularly
  - SSF solar arrays will use sliprings to carry tens of kWe
- The Remote Manipulator System on SSF will be much like a tether crawler
  - exception is that it uses power rather than providing it
  - crawls along SSF truss, stopping periodically to plug into electrical outlets
- Technology demonstration in 1991 on Tethered Satellite System (TSS) Shuttle flight
  - conductive tether with plasma contactors for electrodynamic experiment

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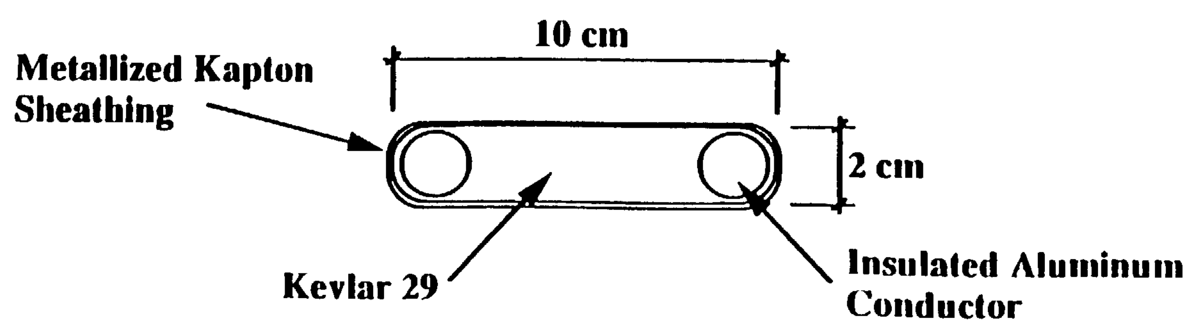
Conductive Tether Properties

- Kevlar 29
- 1.2×10^6 N stress
- Safety factor = 1.5 (using 3 tethers)
- (3) 180m tethers
  - 161m nominal separation
  - 56m radius to transfer habitat
- "Ribbon" shaped cross section
  - to avoid entanglement during reeling/unreeling cycles
  - easier crawler operations
  - 20 cm^2 cross-sectional area
  - radiates conductor heat better due to increased surface area



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g_a Cryo/AB Vehicle Features

- Nominal spin rate = 4 rpm (56m to create 1g)
- Conductive tether
- Sun-tracking solar arrays
- "Crawler" contingency for crew transfer from end to end
- Nominally 4 spin-up/spin-down cycles (1 for conjunction class mission)
  - Outbound - MTV aerobrake and propulsion as countermass
  - Inbound - Empty MTV propulsion and aerobrake as countermass (MEV expended)
    - MTV aerobrake not required because of ECCV; however, it is useful as a countermass and will be retained for fully-reusable mission modes

g_a Cryo/AB Tether Deployment Scenario

This chart outlines the vehicle deployment scenario. Omitted from this chart, for the interest of simplicity, are mid-course corrections, which would follow the same deployment scenario.

g_a Cryo/AB Tether Deployment Scenario

Reference

- Post TMI, RCS fires to separate MTV (propulsion and AB) and MEV (+ transfer habitat) and deploy tether
  - tether slips freely through crawler

- Crawler clamps to CM point on conductive tether to finish deployment
  - 128m tether length outbound
  - 161m tether length inbound

- RCS fires to accelerate end masses to 4 rpm

- Crawler is positioned at CM and deploys solar array and communications laser

- Post Mars arrival, RCS fires to stop rotation

- Solar array/communications laser retract and crawler moves to MEV

- Tether is reeled in, maintaining slight tension

- RCS fires to slow approach to manageable STET speed

- Post berthing, Mars operations commence

- Reverse scenario after TEI using MTV AB and propellant tanks as countermass to transfer habitat

Alternative

- Deploy tether to twice intended length, small ΔV for rotation, then reel tether to nominal length
  - saves propellant, but increases tether mass

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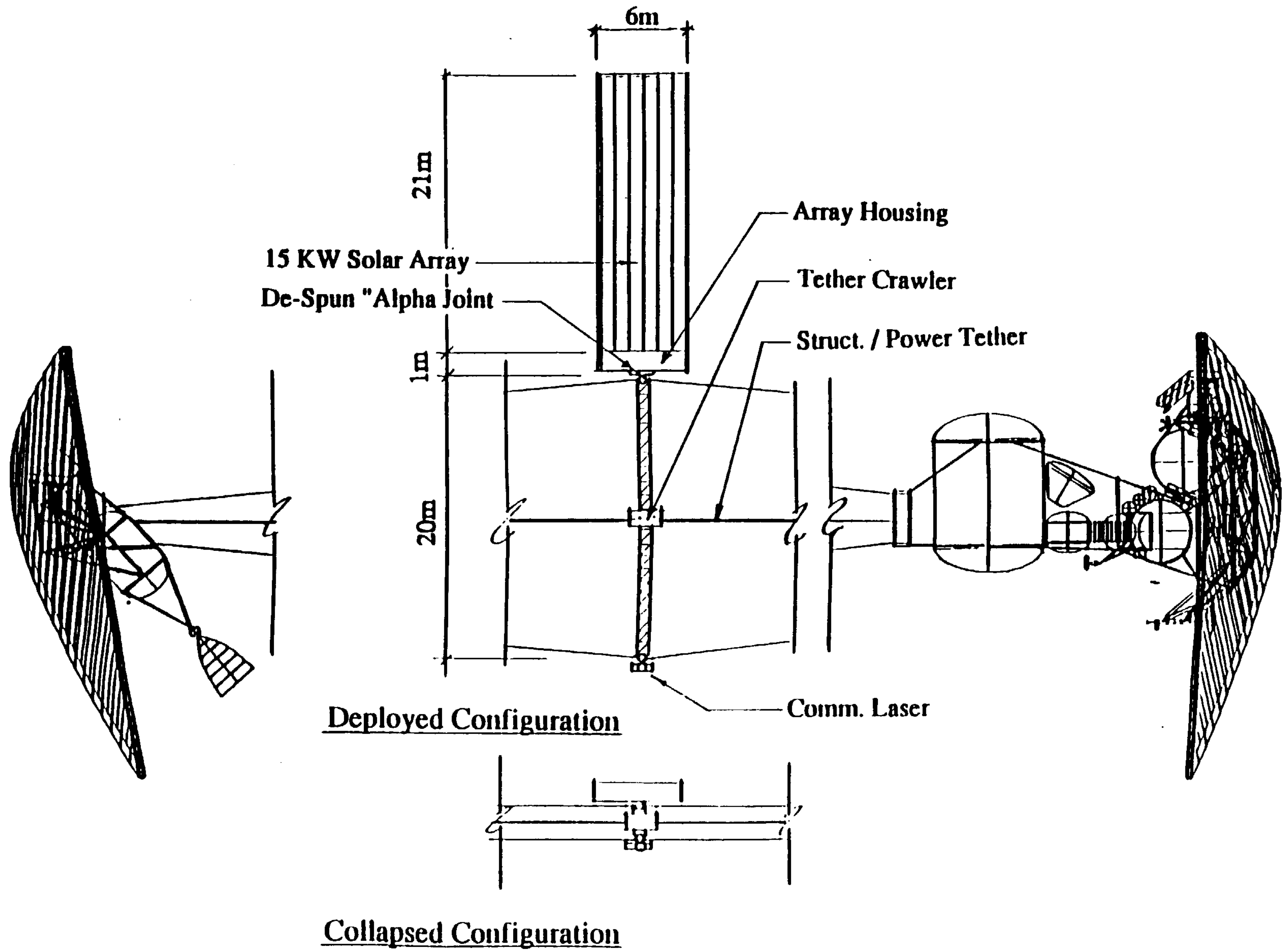
g_a Cryo/AB "Crawler/Mast" Configuration

This chart shows a detail of the central "crawler/mast" in the deployed and collapsed configurations. The solar array and the communications laser deploy on a deployable truss to separate the tethers and form a planar "beam". In the collapsed configuration, the solar array and the communications laser fold-up, spin 90° and package below the transfer hab on the MTV.

g_a Cryo/AB "Crawler/Mast" Configuration



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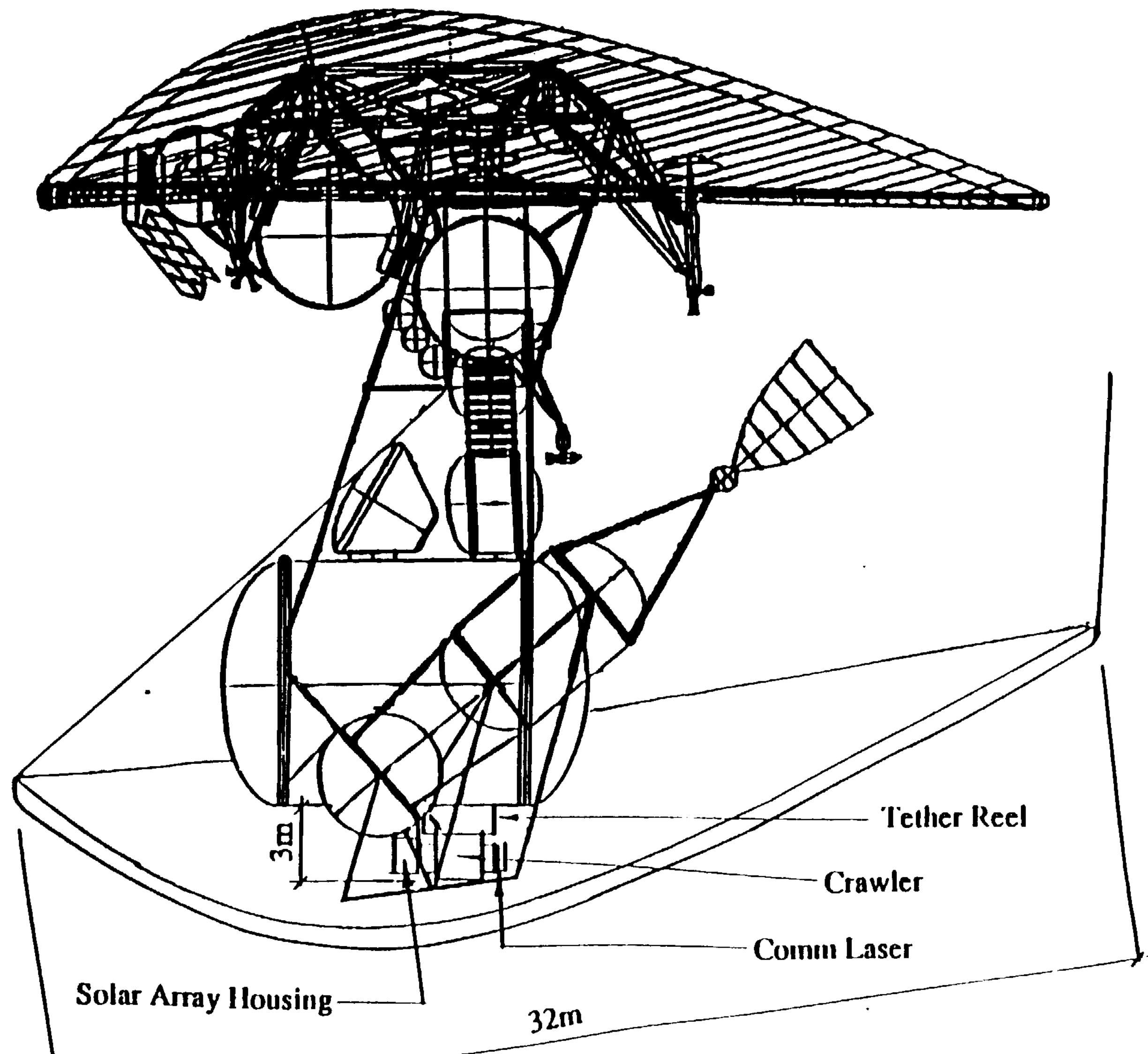
g_a Cryo/AB Packaging Configuration

This chart shows the packaging configuration for the solar array, communications laser, crawler, and tether reel. Due to aerobraking constraints, the MTV aerobrake has to be 2m larger than the MEV aerobrake, which will cause problems in fabrication commonality. The MTV has been designed so that the transfer module and the artificial gravity equipment can slip out to deploy the tether for spin-up.

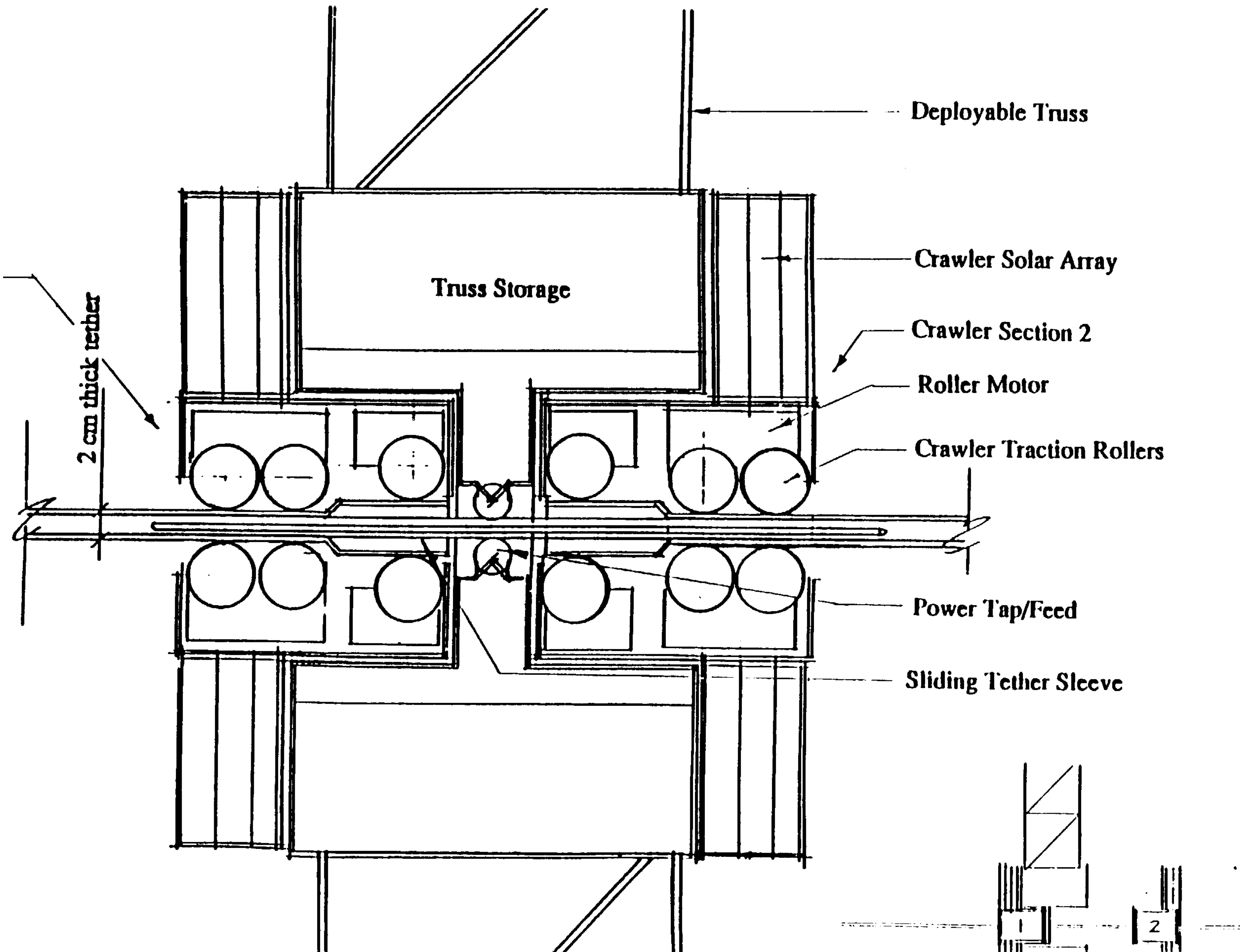
g_a Cryo/AB Packaging Configuration



Tether Crawler Configuration

A detail of the tether crawler is shown on this chart. The crawler is divided into 2 sections so that one section can always be at the CM to support the deployable truss and the tether. The crawler taps into the aluminum conductor to transfer power from the power source to the habitation areas. Each crawler section has 2 small solar arrays for power during movement along the tether and 2 roller motors.

Tether Crawler Configuration



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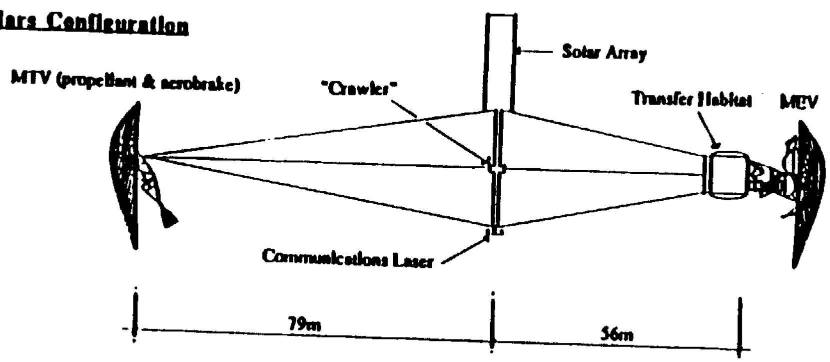
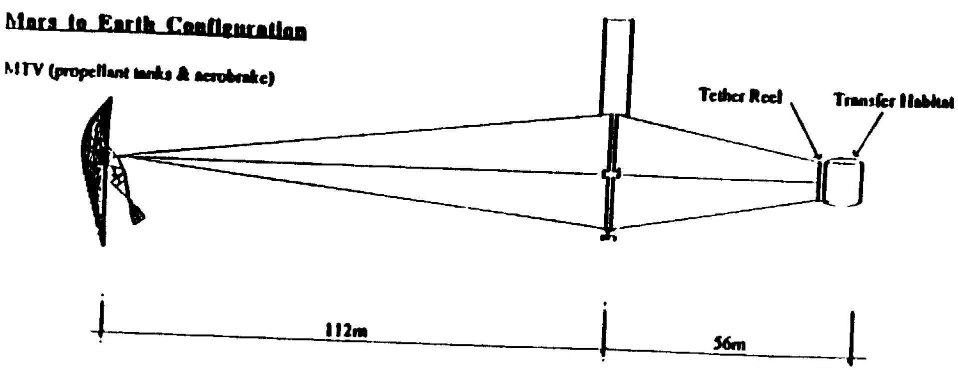
g_a Cryo/AB Mass Statement

Earth to Mars Configuration



Mars to Earth Configuration



Element	*ECCV return*	*Earth Aerocapture*
MTV Mars aerobrake	23758	23758
MTV crew hab module 'dry'	28531	28531
MTV consumables & resupply	7096	7096
MTV science	1000	1000
MTV propulsion stage	21847	27473
MTV MPS propellant load	103129	132177
*MTV Art-g added hardware wt	8057	8057
*MTV Art-g added RCS prop	9004	10627
MTV total	**202422**	**238719**
MEV Mars capture & desc aerobrake	15138	15138
MEV ascent stage	22754	22754
MEV descent stage	21457	21457
MEV surface cargo	25000	25000
MEV total	**84349**	**84349**
ECCV		
Cargo to Mars orbit only	7000	0
MTV-TMI interstage wt	0	0
	500	500
TMI inert stage wt	62820	69080
TMI propellant load	565270	621540
TMI stage total	**628090**	**690620**
IMLEO	**922,361**	**1,014,188**

* *Artifical -g system weight penalties* — *all masses in kg*
Earth aerocapture configuration shown in diagram

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g_a Cryo/AB Penalty Assessment

This chart outlines the penalties of choosing artificial gravity for this particular configuration. The added hardware mass is not as critical as the propellant required to transfer that hardware to and from Mars. The MTV aerobrake is a major penalty in this scenario because it is larger and increases the IMLEO if it is used as a countermass (see previous mass statement). The spin-up/spin-down cycles also greatly increase the system's mass because a spin-down is required for mid-course correction, and, if the MTV aerobrake is retained, an extra 23 t has to be spun-up following mid-course correction. Another penalty of this system is the addition of a despun joint to allow the solar array and communications laser to track the sun and Earth respectively.

g_a Cryo/AB Penalty Assessment

- 15 to 27% added mass
  - (3) tethers
  - Tether reel
  - Tether crawler
  - Added solar array
  - Added communications laser
  - Lock joints for transfer hab
  - Added RCS and propellant
  - Added TMI/TEI propellant

- MTV aerobrake
  - 2 m larger than MEV aerobrake
    - needed due to packaging constraints
    - complicates fabrication due to different sizes
  - Needed for inbound countermass - not needed in 0g option

- Spin-up/spin-down cycles
  - Mid-course correction problems

- "De-spun" joint for power and communication

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Nuclear Thermal Rocket Vehicle Artificial Gravity Configuration

The NTR artificial gravity (g_a) configuration looks exactly like the μg configuration, except longer. A rigid spinner, it uses the crew systems as one end mass and the reactor/engine as the other. The vehicle rotates nominally at 3.98 rpm outbound (56.5 m to create 1g) and 3.83 rpm inbound (61 m to create 1g). The truss used is similar to that in the μg configuration, but actually carries weight in the induced gravity field. The spin radius of the habitation system is practically constant with mission phase because the Earth departure and Mars arrival drop-tanks are located at the vehicle CM. Four spin-up/spin-down cycles are presumed for the nominal mission case.

The NTR vehicle concept is the most amenable to adaptation for artificial gravity of any of the reference vehicles, because its μg and g_a manifestations are so similar. Only a longer (perhaps stronger) truss, added RCS and TMI/TEI propellant, and despun mountings for power and communication systems are required.

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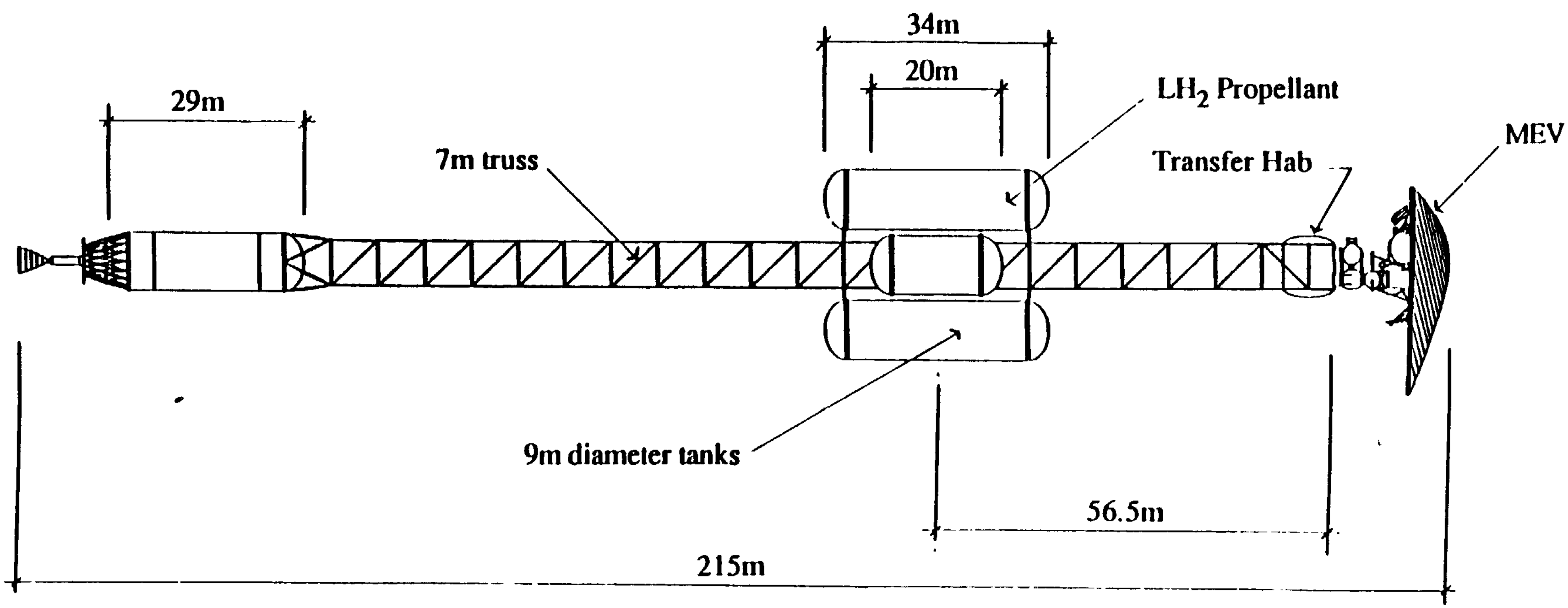
g_a NTR Vehicle Features

- Nominal spin rate
  - 3.98 rpm outbound (56.5 m to create 1g)
  - 3.83 rpm inbound (61 m to create 1g)
- 7 m square cross-section truss
- Earth departure, Mars arrival, and Mars departure tanks placed on CM
  - minimize CM movement when tanks are dropped
- Nominally 4 spin-up/spin-down cycles
  - Earth arrival propellant and engine used as countermass

g_a NTR Configuration

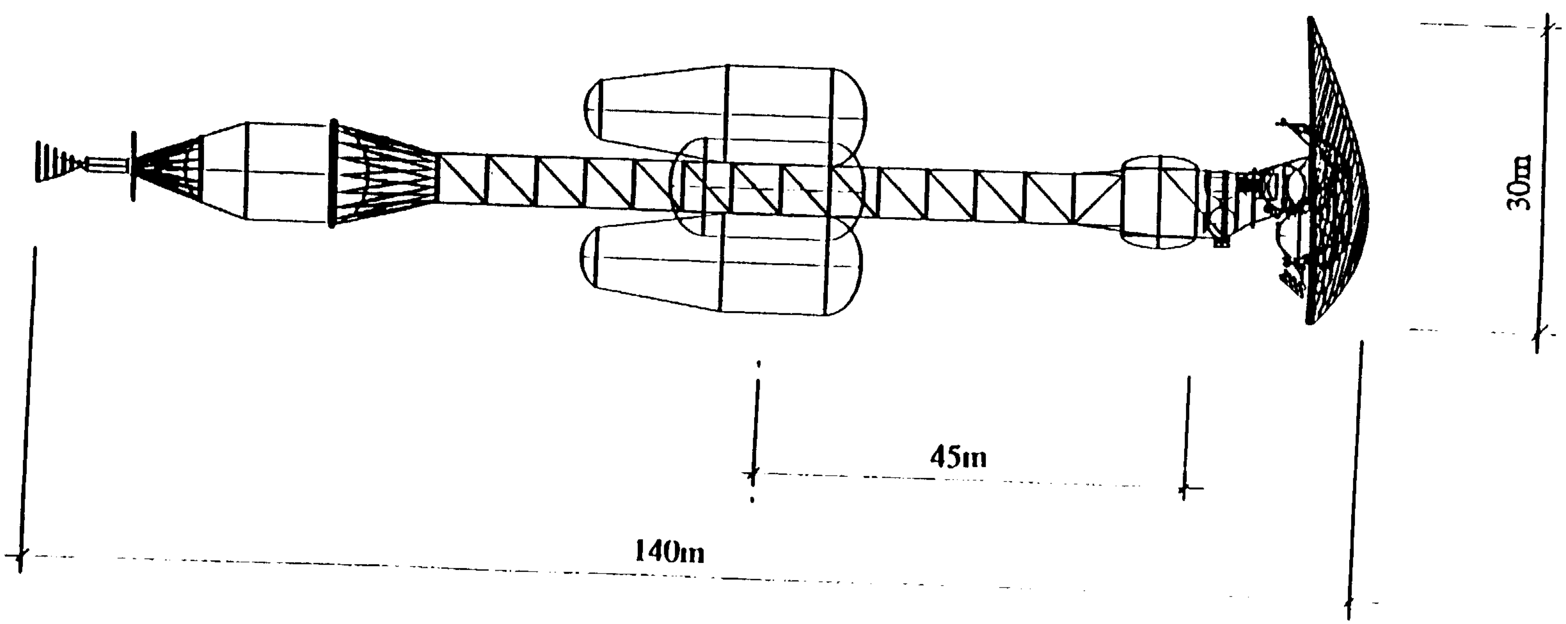
The NTR artificial gravity configuration makes use of the Earth arrival propellant as a countermass and has a rigid truss as connection. The Earth departure, Mars arrival, and Mars departure tanks are located at the CM so, as dropped, minimally disturb the total vehicle CM (4.5m total movement). The Earth arrival propellant is also used as radiation protection for the crew areas from the NERVA type engine.

g_a NTR Configuration



MTV	34.9
MEV	73.1
Prop./Structure	679.3
IMLEO	**787.3 t**

g_a (1/3g) NTR Configuration



MTV	34.9
MEV	73.1
Prop./Structure	643.1
IMLEO	751.1 t

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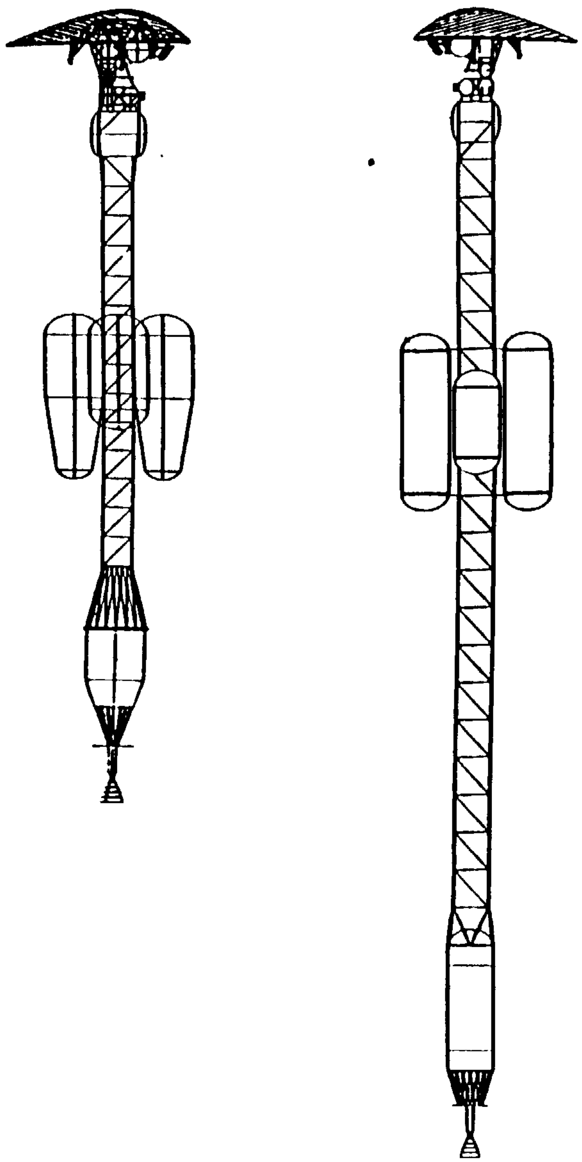
g_a NTR Mass Statement



Element	Mars-g	Earth-g
MEV desc aerobrake	7000	7000
MEV ascent stage	22464	22464
MEV descent stage	18659	18659
MEV surface cargo	25000	25000
MEV total	**73118**	**73118**
MTV crew hab module 'dry'	28531	28531
MTV consummables & resupply	5408	5408
MTV science	1000	1000
MTV crew hab sys tot	34939	34939
MTV frame, propulsion, & shield wt	20033	23031
*MTV added Art-g RCS hardware	650	650
*MTV added Art-g RCS prop	2628	7010
Earth Orbit Capture (EOC) prop	28305	29714
Trans Earth Inject (TEI) prop	60829	64541
EOC/TEI common tank wt	14129	14789
Mars Orbit Capture (MOC) prop	154250	160580
MOC tanks	25967	26788
Trans Mars Inject (TMI) prop	292320	306410
TMI tanks	43891	45719
ECCV	0	0
Cargo to Mars orbit only	0	0
IMLEO	**751077**	**787289**

* Art-g spinup sys wt, in addition to nominal maneuver RCS sys

all masses in kg's

Mac chart: M Art-g 2016 NTR cover pg

2005 conjunction mission NTR vehicle

A NERVA NTR vehicle design was done for a low energy 2005 conjunction mission which is characterized by a 983 day trip time with 482 days at Mars. For this mission a ECCV was taken for crew return to Earth and the vehicle was expended. The engine for this vehicle was the reference NERVA derivative of 9684 kg, Isp of 925 sec and a shadow shield wt of 4.5 t. A nominal crew of 6 was taken. Both MOC propellant and TEI propellant were carried in a single aft tank since the dV for these burns were quite small when compared to the higher dV of opposition trajectories. The payload carried to Mars was a follows:

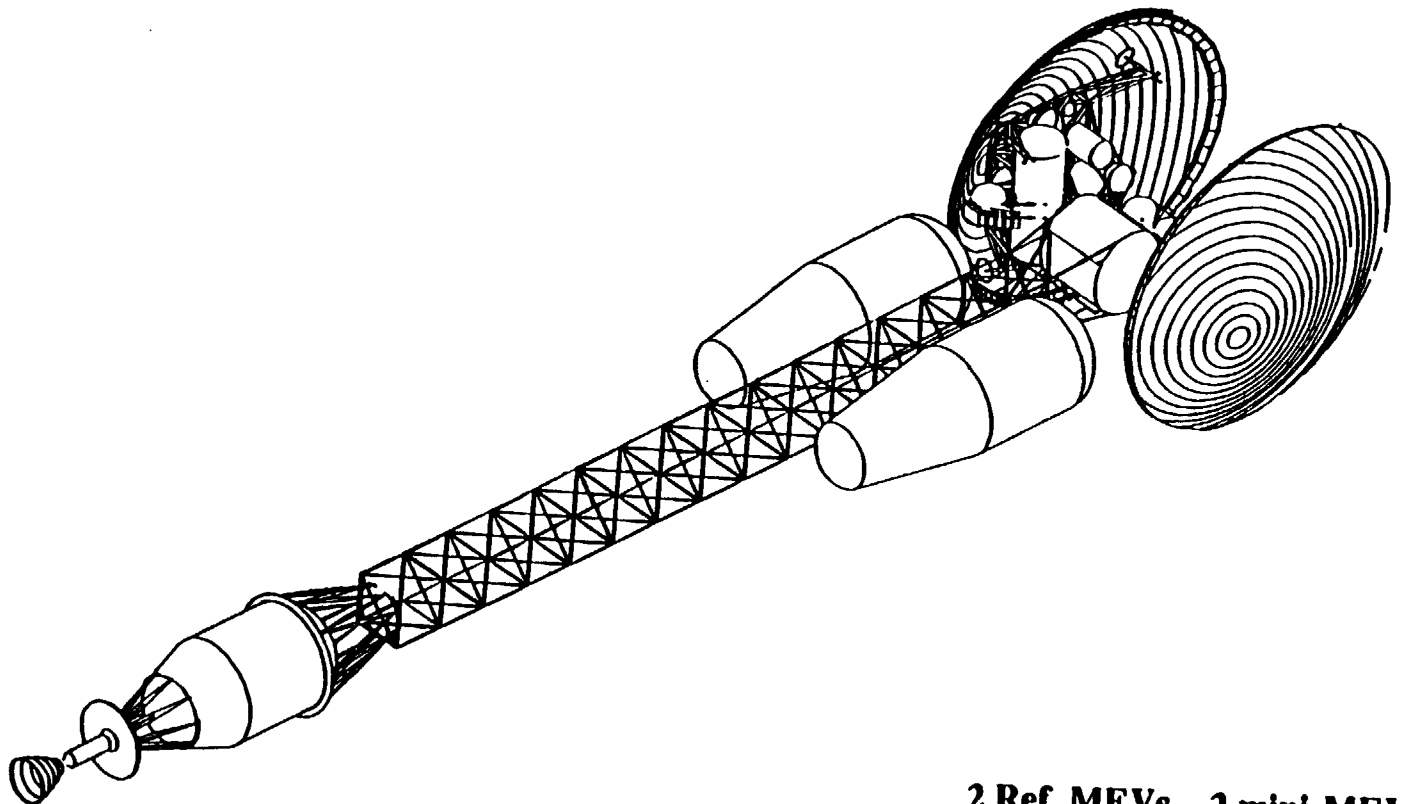
(1) 2 reference MEVs and 30 t of cargo to Mars orbit
(2) 2 mini MEVs and 30 t of cargo to Mars orbit
(3) 2 mini MEVs and 10 t for cargo to Mars orbit

For case (1) the following sensitivities were evaluated:

(1) NTR Isp on IMLEO
(2) Crew size affect on IMLEO
(3) Cargo to Mars orbit affect on IMLEO
(4) Vehicle expended vs vehicle recovered modes

Results are shown on the following two charts

g_a (1/3g) NTR Configuration (2005 Conjunction Mission)



	2 Ref. MEVs 30 t Cargo	2 mini-MEVs 30 t Cargo	2 mini-MEVs 10t Cargo
MEV	146.2	79.5	79.5
Transfer Hab	62.2	62.2	62.2
Prop./Cargo/Str.	396.9	323.5	281.1
IMLEO	605.3 t	465.2 t	422.8 t

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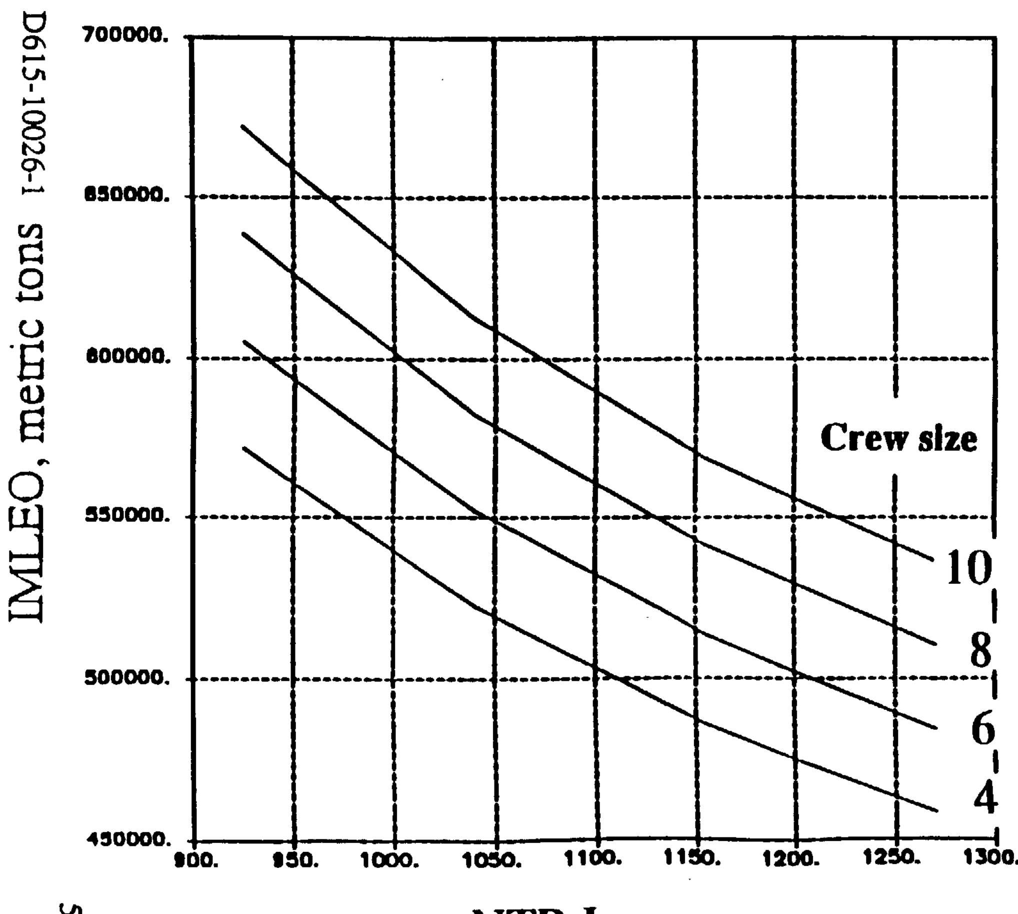
2005 Conjunction NERVA NTR Vehicle Trades

Effect of Crew Size, Eng Isp, Mars Orbit Cargo, & Earth Return Option on Veh IMLEO

983 day trip time, 2 x 73 t MEV's, Art-g (Mars-g) configuration, one 9684 kg 75000 lbf eng, 4.5 t sh....
Nominal case: crew of 6, Isp=925, 30k cargo to orbit, ECCV return, IMLEO=605345 (kg)

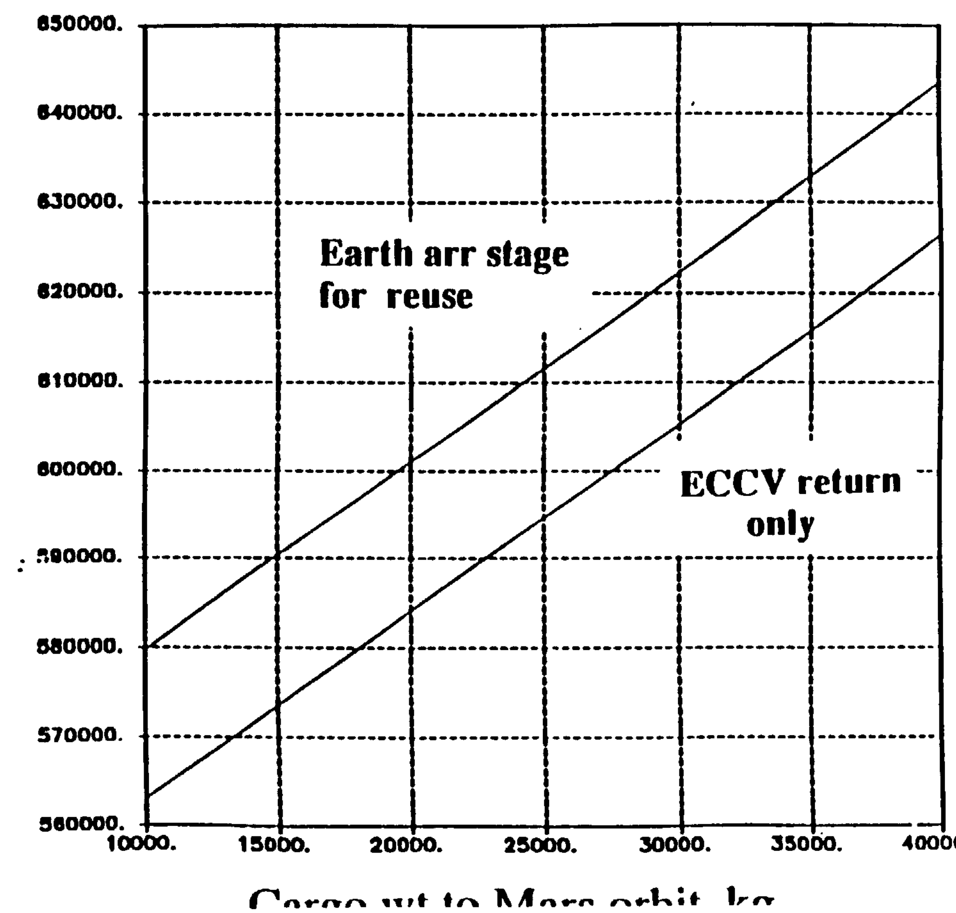
IMLEO vs NTR eng Isp and crew size

- 4 t IMLEO savings per every 10 sec Isp increase
- 17 t IMLEO increase per single crew member addition above 4



IMLEO vs cargo to orbit and Earth return option

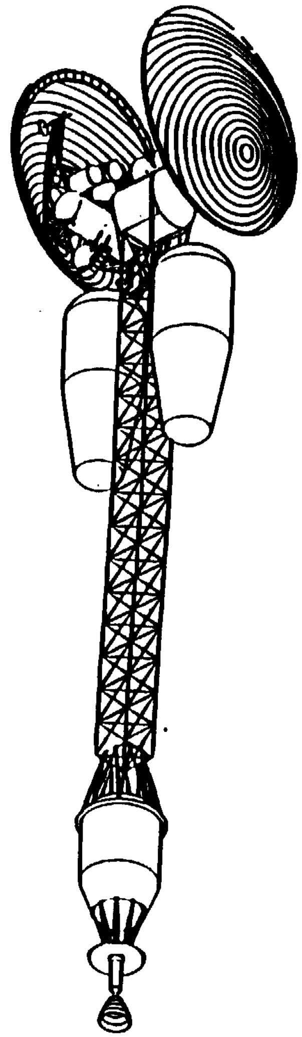
- 22 t IMLEO increase for every 10 t delivered to Mars orbit
- 16 t IMLEO increase to recapture veh in 500 km by 24 hr Earth orbit for reuse vs ECCV return w veh expended



Art-g (Mars-g) NTRVehicle for 2005 Conjuction Mission

Nominal payload: 2 x 73t MEV's (desc only aeroshell), 30k science cargo to mars orbit only, ECCV crew return, no vehicle reuse, Crew of 6, 983 day trip time

TMI dV: 4267 m/s (includes 300 m/s g-loss), MOC dV: 863 m/sec, TEI dV: 1179 m/s 6/7/90



Element	*2 Ref MEVs 30k cargo*	*2 mini MEV's 30k cargo*	*2 mini MEV's 10k cargo*
MEV desc only aerobrake	7000		
MEV ascent stage	22464		
MEV descent stage	18659		
MEV surface cargo	25000		
MEV total	73118	39752	39752
x 2	146236	79504	79504
MTV crew hab module 'dry'	34775	34775	34775
MTV consumables & resupply	19398	19398	19398
MTV crew habit module total	54173	54173	54173
MTV frame,propulsion & shield wt	20033	20027	20027
*MTV Art-g added RCS hardware	650	650	650
*MTV Art-g added RCS prop	6348	5134	4588
Trans Earth Injec (TEI) prop	19648	19185	19042
Mars Orbit Capture (MOC) prop	35166	26955	24470
MOC/TEI/EOC common tank wt	9669	8543	8202
MTV propulsion/frame/prop total	91514	80494	76979
TMI tanks wt	36911	29732	27563
Trans Mars Injec (TMI) prop	238510	183220	166520
TMI stage total	275420	212950	194083
ECCV	8000	8000	8000
MTV Cargo to Mars orbit only	30000	30000	10000
IMLEO	605343	465121	422739

* Art-g spinup wt additions

Mac chart: M NTR 2005 Conj cover pg

Crew hab mod - MTV for 2005 Conj Art-g (Mars-g) NTR Veh

Art-g [Mars g], Crew of 6, 983 day total trip time 6/5/90

		Element	mass (kg)	Rationale
Crew hab module dry weight	[360]	Structure	10878	Cyl length: 9 m, dia: 7.6 m, ellip ends, 3 levels; tri grid w beam supports. Tens. ties
	[363]	ECLSS	5743	SSF derived with same degree of closure. supports a crew of 6 for 983 days
		Command/Control/Power		
	[364]	• Internal	1159	ECWS, DMS, batteries, other avionics/computing/monitoring eq, conditioning equip
	[i281]	• External Power	1539	Solar array,boom, power distribution,power management,fuel cell system
	[368-316]	Man systems	4457	Wts -all sys:SSFderived(as a funct. of crew size&occupancy time)for Mars missions
	[316]	Crew & effects	660	110 kg per person including personal belongings
	[373]	Spares/Tools	1803	Subsys component level spares. Life crit sys are 2 fault tolerent (approach of SSF)
	[247]	Radiation shelter	2439	Provides 10 g/cm2 protection + 3-5 g/cm2 provided by vehicle structure and equip
	[377]	Weight growth	3705	15% weight growth for dry mass excluding crew & effects and radiation shelter
	[237]	Airlocks	1530	2 x 765 kg external airlocks (shuttle type airlocks modified for MTV mission)
	[330]	EVA suits	0	EVA suits weight counted in MEV ascent cab weight statement
	[i60]	TTNC & GN&C platforms wt	<u>862</u>	3 platforms
	[378]	***MTV 'dry' crew hab mod wt***	34775	'dry' hab module represents structure and support systems equip & hardware that are dependant on crew size and independant of mission duration
Consumables and resupply	[371]	*On board equip resupply	3605	Based on adjusted SSF resupply reqts for pot w, hyg w, ARS,TCS/THC &WMS
	[398]	*Consumables	<u>15793</u>	Crew of 6 for 983 days; food:2.04 kg/man/day, food pkg:0.227, pharmaceuticals: 0.25 other: 0.291 Clothes: 42 kg/man. food vol: 0.0055 m3/man/day, other: 0.0018.
	Sum	***MTV crew mod 'wet' wt***	19398	
	[i65]	*Transfer science equipment	0	Inb and outb MTV science hardware and supplies
	[i57]	Remote Manipulator-arm Sys	0	all large external self assembly hardware left in LEO
	124-230	***MTV crew mod & support systems weight***	54173	This wt refects the Boeing ref crew module subsystems modified for an additional 2 crew members and a much longer mission duration. The mod 'dry' wt represents a SSF type closed ECLS Sys (air >99%, water >95%) that serves the crew with 2 fault tolerence on all life critical sys except structure. Its wt varies primarly with crew size. consumables wt varies with crew size and mission duration.

* *MTV hab mod consumables, resupply, and transit science dependant on mission duration, and free abort requirement. i.e. crew mod 'wet' wt will vary for different missions*

Mac chart: M NTR 2005 Conj mod Art-g
synthesis model run# marsntrmtv dat:178

2005 Conj Art-g (Mars-g) NERVA NTR Vehicle:

Frame, propulsion sys, Artificial-g sys, & shield wt: Isp = 925 6/5/90

	Element	*mass (kg)*	*Rationale*
[159]	Spacecraft frame (truss) struct and aft tank struts wt	3480	Truss struct: Graphite epoxy, Ec= 16 Msi, Den=0.06 lb/in3, 5 m by 70 m SSF type
[183]	Maneuver RCS inert	1300	Systems fore and aft that fire simultaneously to provide rotation about vch Cg
[709]	Main prop line wt art-g	451	Main line from tank lines to reactor, L=40 m,d wall s steel;dens=7833kg/m3,t=0.8mm
[160]	Mass growth	785	15% mass growth
[518]	Engines wt (1)	9684	*75k lbf Thrust,* wt estimate: Stan Borowski of NASA/LeRC
[543]	Engine shield wt (1)	4500	Shadow shield wt from NASA/LeRC propulsion task order
[118]	Maneuver RCS prop wt	483	MTV maneuver RCS dV = 20, Isp = 400, gaseous O2/H2
[696]	***Frame & propul 'dry' wt***	20683	
[758+759]	Outb 1st spinup/down prop	2184	Outbound leg, 4 RPM Mars-g spin untill spindown for outb midc correction burn
[756+757]	Outb 2nd spinup/down prop	2175	Outbound leg, from midc correction to Mars capture
[754+755]	Inorbit spinup/down prop	846	Inorbit spinup
[752+753]	Inb 1st spinup/down prop	572	Inbound leg, 4 RPM Mars-g spin untill spindown for Inb midc correction burn
[750+751]	Inb 2nd spinup/down prop	570	Inbound leg, from midc correction to Earth capture
[761]	***Total Art-g spinup/down prop***	6347	Total RCS prop for all 5 Art-g spin maneuvers

Mars dep & Earth capt stages - 2005 Conj Art-g (Mars-g) NTR Veh

14% tank fraction, 3% cooldown penalty, 2% prop margin, Isp = 925 sec 6/5/90

		Element	Mass (km)	Rationale
Mars dep	[128]	Mars dep usable prop load	13220	***Mars dep dV= 1179 m/s; eng Isp=925 sec,*** H2 density =70.8
	[703]	Mars dep prop residuals	264	2% residuals/reserve left after boiloff,burn and cooldown
	[699]	Mars dep burn 'cooldown' prop	396	3% post burn prop for reactor cooldown; no thrust/Isp counted in this approximation
	[498]	Mars dep stg outbound boiloff	2293	Out b boiloff for given MLI & VCS insul.;no refrig,based on Boeing 'CRYSTORE'
	[545]	Mars dep stg inorbit boiloff	2239	482 day inorbit stay time
	[122]	Inbound midcourse prop	1235	Inb midc maneuver dV=90 m/s; done by main propulsion system
	[711]	**Tot Mars dep stg prop load**	**19648**	total at time of TMI burn
Earth capt	[561]	Earth arr stg usable prop tot	0	***Earth arr dV=0.0m/s; propulsive burn capture into 500 km by 24 hr ellip orbit***
	[704]	Earth arr stg prop residuals	0	2% residuals/reserve left after boiloff,burn and cooldown
	[700]	Earth arr stg 'cooldown' prop	0	3% post burn prop for reactor cooldown; no thrust/Isp counted in this approximation
	[562]	Earth arr stg outbound boiloff	0	983 day boff period;added boff from this tank also accounted in M capt &M dep boff
	[563]	**Total Earth arr stg prop load**	**0**	Total at time of TMI burn
	[570]	**Total combined prop load in aft tank**	**54811**	Mars capture/M dep prop: put in 1 tank; (along veh centerline aids NTR radiation attenuation)
Common tank				***1 continuous reinforced Silicon Carbide/Al metal martrix tank:*** dia:10 m, L:11.5 m,
	[683]	Single M dep/E arr tank wt	4142	filament wound; dens= 2436 kg/m3; 37ksi Wk. stress; tank skin thickness = 4.0 mm
	[684]	MLI wt	1369	MLI: density = 32 (kg/m3); 200 layers at 20 layers/cm. wt=SA x no. layers x dens
	[685]	Vapor cooled shield wt	1081	2 VCS - 2 x 0.13 mm Al sheets with 0.57 kg/m2 honeycomb core each
	[686]	Meteoriod shield wt	918	One 0.80 mm sheet of Al; comparsion: SSF plans 0.8 mm, Mariner 9 used 0.4 mm
	[687]	Propel line/valves wt	225	length =10 m,double wall stainless steel H2 prop line; density= 7833 kg/m3, t=0.8mm
	[689]	Mass growth wt	1934	25% wt growth for tank shell, MLI,VCS,meteor shield,prop line & attachment
	[565]	Sum of inerts:single tank	9669	Total for single tank with all tank related inerts.
	[566]	Total for 1 tank	9669	Overall tank fraction [571] = 15.0 %
	[572]	***Combined Mars capt & Mars dep & Earth arr tank set & prop load***	**64480**	Total for ' Mars capture/Mars dep/Earth arr tank set ' at time of TMI burn
	[171]	***IMLEO***	**605340**	

Earth dep stg & Mars capt prop - 2005 Conj Art-g (Mars-g) NTR Veh

14 % tank fraction, 3% cooldown penalty, 2% prop margin, Isp = 925 *6/5/90*

	[586]	Earth dep usable propel tot	227150	***Earth dep dV: 4267m/s (includes 300 m/s g-loss); Isp = 925 sec***
	[701]	Earth dep prop residuals	4543	2% residuals/reserve left after boiloff, burn prop, and cooldown
	[697]	Earth dep burn 'cooldown' prop	6814	3% post burn prop for reactor cooldown: no thrust/Isp counted for this estimated %
Earth dep stg wt	[705]	**Tot Earth dep stg prop load**	**238507**	

				2 continuous reinforced Silicon Carbide /Al metal matrix tanks: dia: 10.0 m,
	[668]	Single tank wt (cyl/ellip ends)	7834	L:24.1 m, dens= 2436 kg/m3; 37ksi wk. stress, thickness = 4.0 mm, root 2 ellip ends
	[669]	MLI wt	2589	MLI: density = 32 (kg/m3); 200 layers at 20 layers/cm. wt=SA x no. layers x
	[670]	Vapor Cooled Shield wt	2045	density2 VCS: at 2 x 0.13mm Al outer sheets with 0.57 kg/m2 honeycombcore
	[671]	Meteoriod shield wt	1736	eachOne 0.80 mm sheet of Al; comparison: SSF uses 0.8 mm, Mariner 9 used 0.40
	[1312]	Tank/frame attachment	400	mmTank attachment mounting brackets & hardware as well as tank release
	[672]	Tank feed prop line wt	159	mechanism Short prop line from tank to main prop line:double wall, stainless steel:
	[674]	Mass growth wt	3691	10 meter 25% wt growth for tank inert,MLI,VCS,meteor shield,proplines, tank/veh
	[588]	Sum of single tank inerts	18455	attachment Total single tank inert wt;
	[592]	Total for 2 tanks	36911	Total for ' Earth dep tank set ': inert wt; Overall tank fraction [593] = 13.7 %
	[597]	***Earth Dep stage tot wt***	275418	Total Earth dep stg weight at time of Trans Mars Injection burn

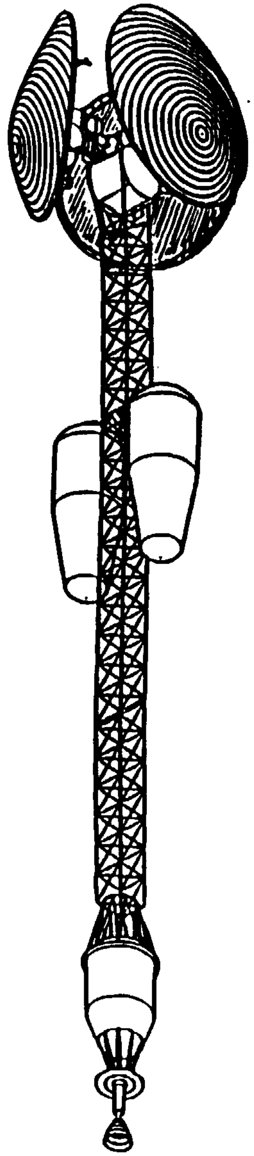
[610]	Mars arr usable prop tot	28816	***Mars arr dV: 863 m/s; eng Isp=925,*** H2 density = 70.8
[702]	Mars arr prop residuals	576	2% residuals /reserveleft after boiloff, burn prop, and cooldown
[698]	Mars arr burn 'cooldown' prop	864	3% post burn prop used for reactor cooldown; prelim;based on Westingh. estimate
[611]	Mars arr stg outbound boiloff	1673	Boiloff for given MLI,VCS and Outb trip time; based on Boeing's 'CRYSTORE' prog
[121]	Outbound midcourse prop	3235	Outb midc maneuver dV = 120 m/s; done by main propulsion from M arr tanks
[612]	**Tot Mars arr stg prop load**	**35164**	

	[633]	Single M arr tank wt	0	***0 SiC/Al metal matrix tanks;*** dia:10.0 m, L:0.0 m, dens=2436 kg/m3;thick=4.0mm
	[634]	MLI wt	0	MLI: density = 32 (kg/m3); 200 layers (4 inches) at 20 layers/cm
Mars arr stg tank	[635]	Vapor cooled shield wt	0	2 VCS: at 2 x 0.13 mm Al sheets with 0.57 kg/m2 honeycomb core each
not used	[636]	Meteoriod shield wt	0	One 0.80 mm sheet of LiAl: assumption-LEO assembley in protective hanger
	[1312]	Tank/frame attachment	0	Tank attachment mounting brackets & hardware as well as tank release mechanism
	[637]	Tank feed prop line wt	0	Double wall, stainless steel 10 meter H2 propellant line;dens= 7833 kg/m3, t=0.8mm
	[639]	Mass growth wt	0	25% wt growth for tank inert,MLI,VCS,meteor shield,prop line&tank/veh attachment
	[614]	Sum of single tank inerts	0	Total for single tank, with all inerts.
	[615]	Total for 2 tanks	0	Overall tank fraction [620] = 14.5 %
	[617]	***Mars Arr stage wt [not used] Mars capt prop in single aft tank w M dep & E arr prop***	0	

Exploration Emphasis Three Lander Art-g Conj Class NTR ven

20 t or 1 t to Mars orbit, 1/3 g, crew of 6 (MEVs land 4), ECCV ret, Ref NERVA: Isp=925

dV's: TEI = 3900 m/s, MOC = 1530 m/s, TEI = 920 m/s, EOC Vhp = 5525 m/s, midcourse correction burns: outb=40 m/s, inb=40 m/s



Crew return via ECCV, no vehicle reuse

	Element	MEV type: P/L taken & left in Mars orbit:	Ref 20 t	Ref 1 t	Mini 1 t
[1313]	MEV descent only aerobrake		7000	7000	6000
[63]	MEV ascent stage		22464	22464	37366
	MEV descent stage		18659	18659	**n/a
[166]	MEV surface cargo		25000	25000	5000
[1339]	***MEV total***		**73118**	**73118**	**48366**
[106]	***x 3***		**219350**	**219350**	**145098**
[378]	MTV crew hab module 'dry'		34790	34790	34790
[382]	MTV consumables & resupply		18270	18270	18270
[165]	MTV transit science		0	0	0
[381]	***MTV crew habitat module total***		**53060**	**53060**	**53060**
[1356]	Payload taken & left in Mars orbit		20000	1000	1000
[158]	MTV stg frame, struts, & misc inerts		4521	4521	4521
[792]	MTV stg RCS hardware & tankage		2034	1991	1834
[518]	NTR total engine wt		9684	9684	9684
[543]	NTR radiation shadow shield wt		4500	4500	4500
[118]	MTV nominal maneuver RCS prop		503	500	470
[761]	*MTV Art-g RCS spinup/down prop		7216	6851	5539
[121]	Outbound midcourse correction prop		1889	1784	1345
[122]	Inbound midcourse correction prop		431	428	402
[765]	Earth Orbit Capture (EOC) prop		n/a	n/a	n/a
[769]	Trans Earth Injection (TEI) prop		14280	14202	13404
[770]	Mars Orbit Capture (MOC) prop		72066	68104	51544
[657]	MOC/TEI common aft tank wt (1)		14564	14026	11711
[771]	***MTV propulsion/frame/propel tot***		**131690**	**126590**	**10450**
[705]	Trans Mars Inject (TMI) prop		274530	259430	196520
[592]	TMI side tanks wt (2)		41582	39622	31456
[597]	***TMI stage total***		**316110**	**299050**	**227980**
[230]	ECCV		8000	8000	8000

* Art-g prop for 5 spinup/spindown maneuvers

** single stage veh: aeroshell & landing ...

g_a NTR Penalty Assessment

This chart outlines the penalties of using artificial gravity with the NTR configuration. The penalties for this concept are much less than the Cryo/AB version. There is a limited mass increase because the only major difference is the longer truss and despun joints, which subsequently increases propellant loads.

g_a NTR Penalty Assessment

- 7 % added mass
  - Heavier truss (vehicle "hung" from center of rotation)
  - Longer truss
  - Added RCS and propellant
  - Added TMI/TEI propellant

- Spin-up/spin-down cycles
  - Mid-course correction problems

- "De-spun" joints for power and communication

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Solar Electric Propulsion Vehicle Artificial Gravity Configuration

The solar electric vehicle (SEP) artificial gravity (g_a) concept presents complications not present in the lower-performance propulsion concepts. For full-fledged g_a conditions, EP vehicles pose the problem of spinning while thrusting. [An alternative, operational solution may be to fly μg for most of the trajectory, spinning only during the midflight coast intervals (25 to 60 days) and upon arrival at Mars. For STCAEM purposes, however, it is essential to pursue the outcome of a vehicle required to provide artificial gravity for the entire flight.] Because the thrust vector must average tangential to the flight path, the fundamental configuration trade-off is between rotating, high-power transfer assemblies (for the spin vector normal to the ecliptic) and spin-vector precession (for any other orientation).

Of the many possible configuration options identified by STCAEM, the one was chosen that is similar both to the μg SEP and to the NEP g_a concept. This configuration concept, called an *eccentric rotator*, avoids tethers, complex extendible booms or deployable trusses. All components are rigid and the design is simple.

The fundamental concept is that the large solar array is split in two, leaving a gap or slot within which spins a rigid boom supporting the habitable systems. The optimal shape of the two solar array halves has not yet been determined. A single, double-ended slipring assembly (which transmits only habitation-system power levels) is used to despin the vehicle bus. No deployment mechanism is required to change the habitat system separation when the MEV mass is lost. Instead, the rotation rate is adjusted to provide 1g in the center of the long-duration habitat, *according to the habitat's actual separation from the current vehicle mass center*, which shifts after MEV operations. Thus the mass center is not necessarily axially aligned with the engine outrigger or geometric center of the solar array, although it always remains at the zenith relative to the habitat floors. When the mass center is not along the outrigger axis, the outrigger and solar array also orbits the mass center. The engine assemblies therefore trace out circles as they thrust, although the thrust vector orientation remains fixed. For low-thrust systems in particular, this is expected to cause no problems. The solar array, main structure and engine assemblies are used as the countermass to the crew systems.

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g_a Solar Electric Propulsion Vehicle

The next 7 charts show the options being considered for the SEP artificial gravity configuration. The SEP is in the preliminary stage of trading configurations, and one concept has not yet been chosen.

Artificial Gravity for Continuous Thrust Systems

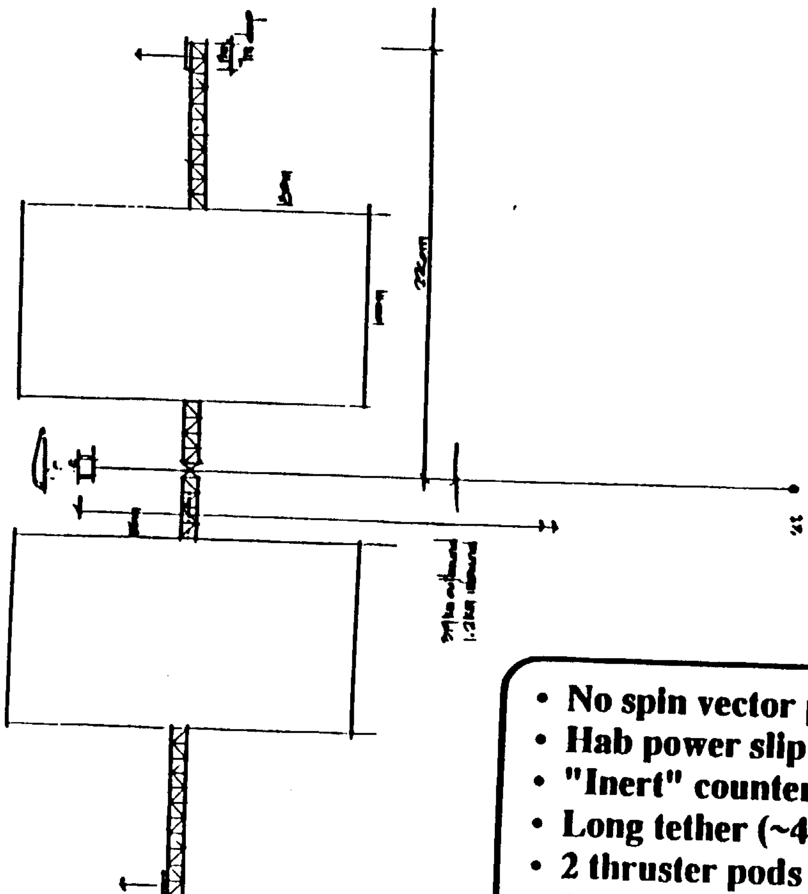
- SEP vehicles lack obvious countermass:
  - inert mass trades with long tether
  - using array as countermass requires heavier array structure

- Spinning while thrusting poses configuration complications:
  - high-power sliprings (single-point failure route)
  - cross-product engines (10 - 15% mass penalty)

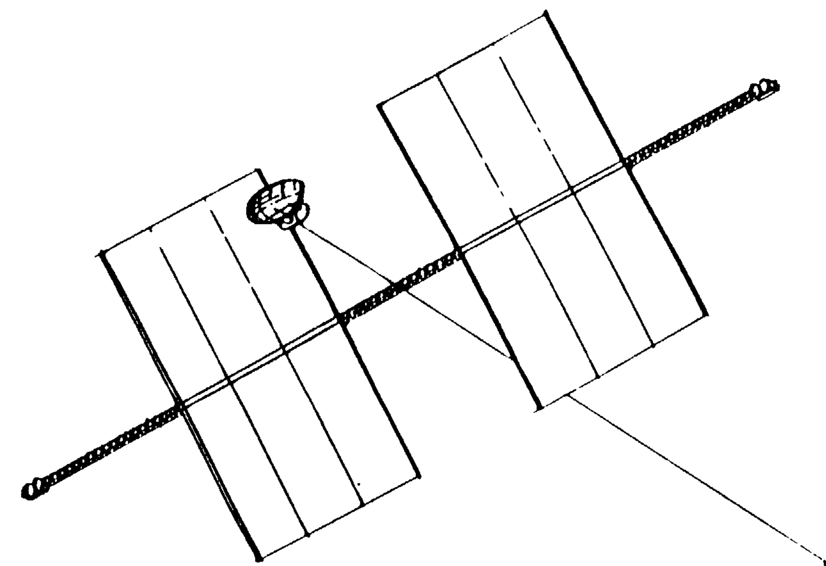
- Least-mass, least mechanically complicating solution
  - 0 g for most of trajectory
  - g_a (no precession) for last 15d before Mars arrival (re-conditioning period)
  - g_a possible for mid-course no-thrust interval (25-60d)
  - full conditioning not perfomance - driven for Earth return

g_a Solar Electric Propulsion Vehicle Option 1

Orthogonal View



Isometric View



- No spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~4 km)
- 2 thruster pods
- 2 solar arrays
- Slipring separating 1 thruster pod and 1 array
- 450 m truss

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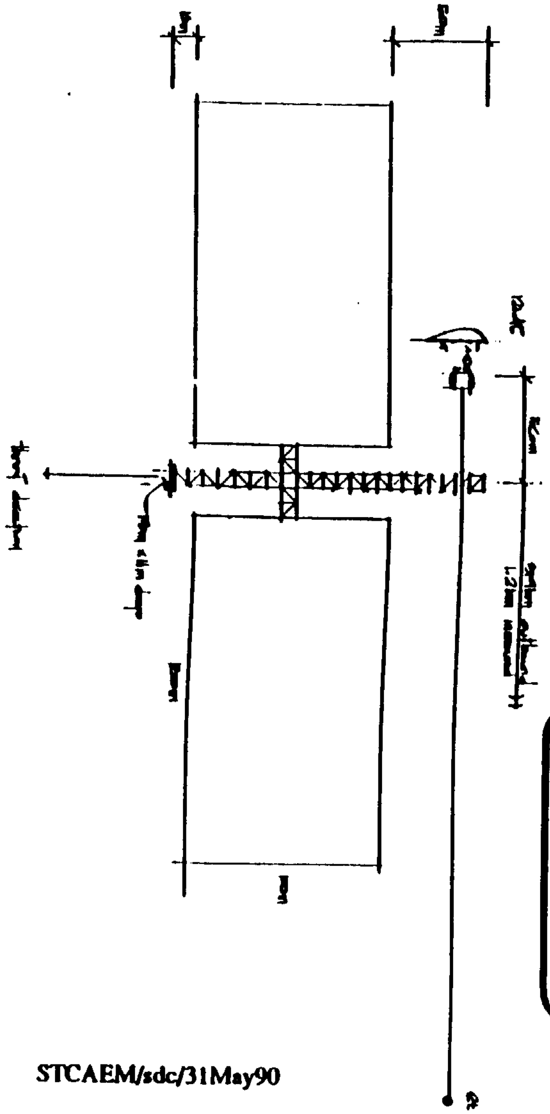
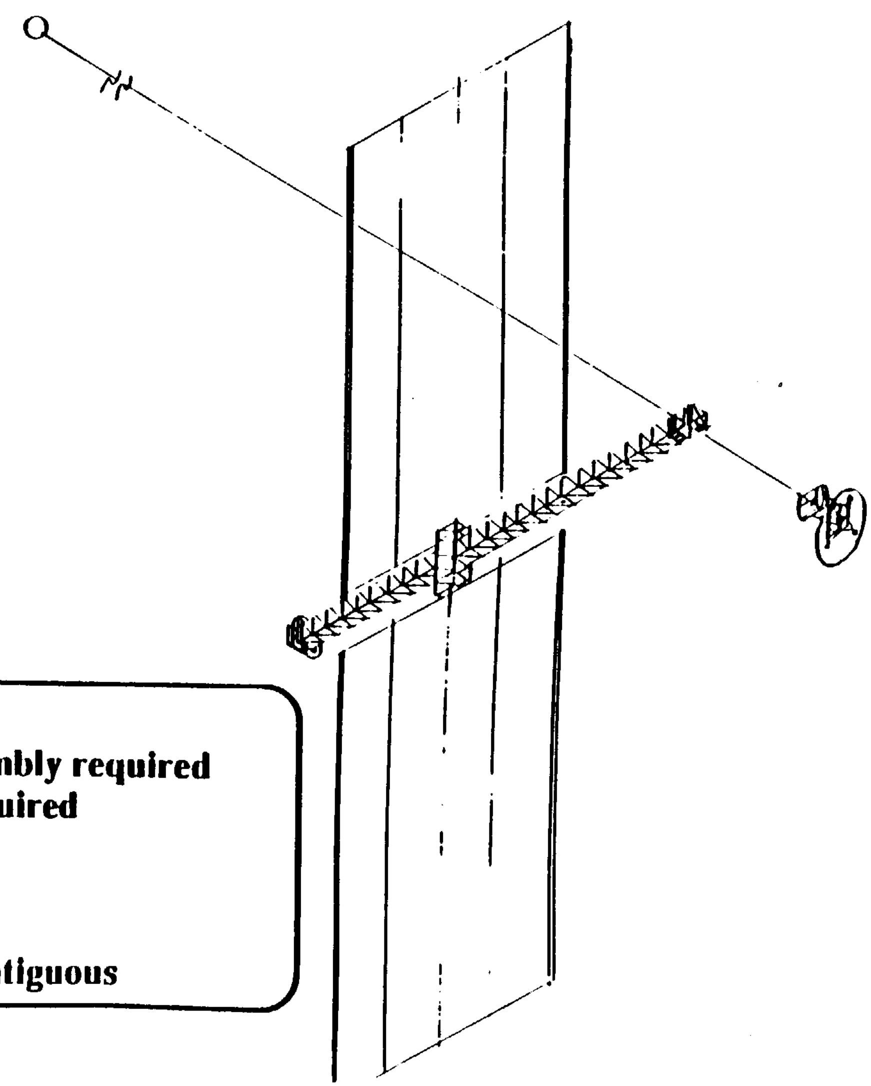
8a Solar Electric Propulsion Vehicle Option 2A

Orthogonal View



Isometric View



- Spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~4 km)
- Single thruster pod
- 2 solar arrays
- Thrusters and arrays contiguous

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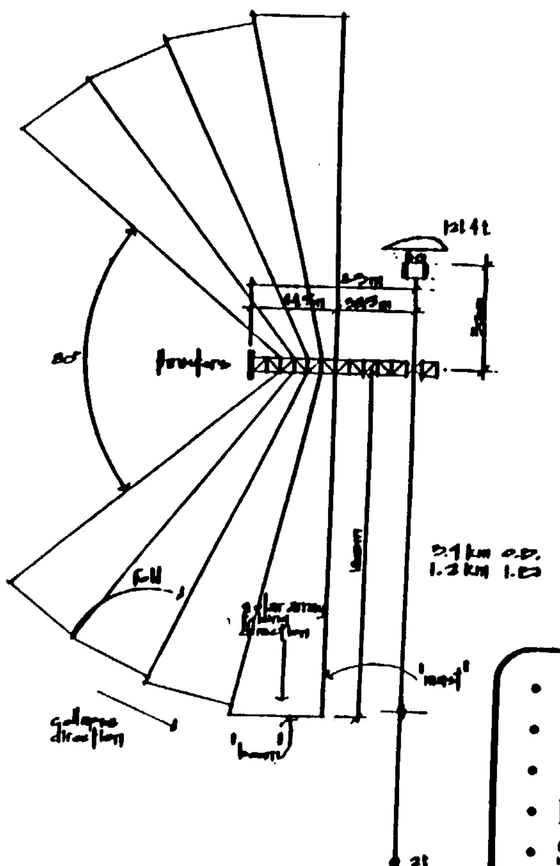
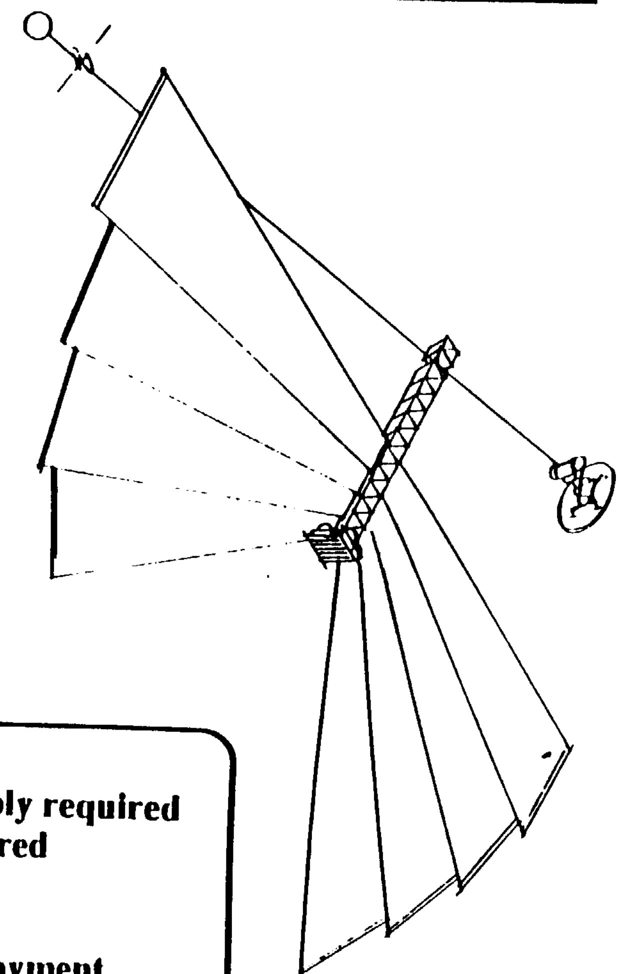
g_a Solar Electric Propulsion Vehicle Option 2B

Orthogonal View



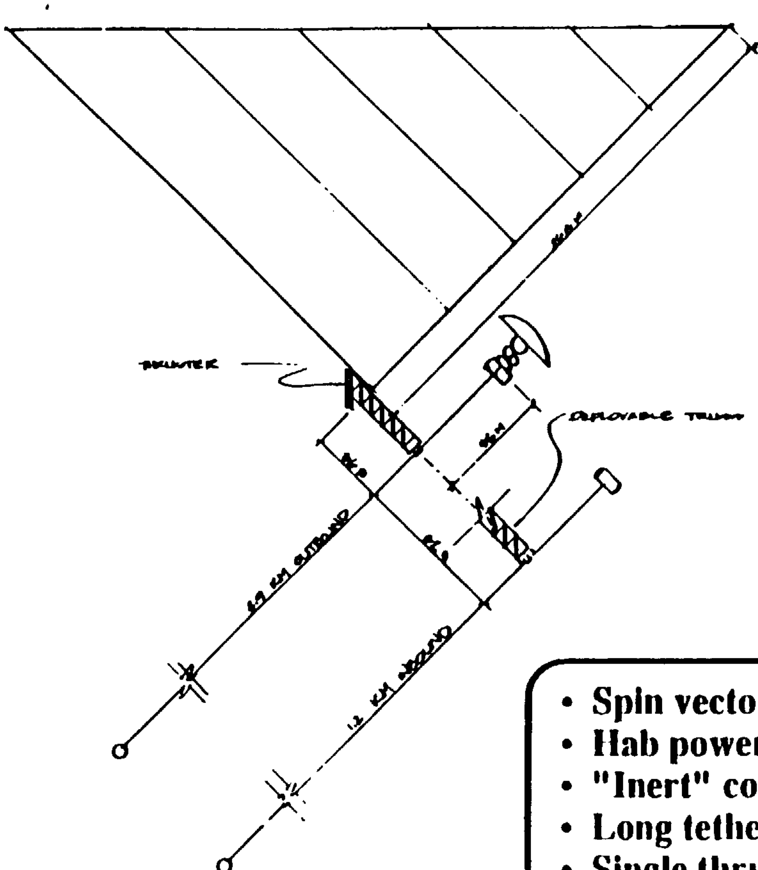
Isometric View



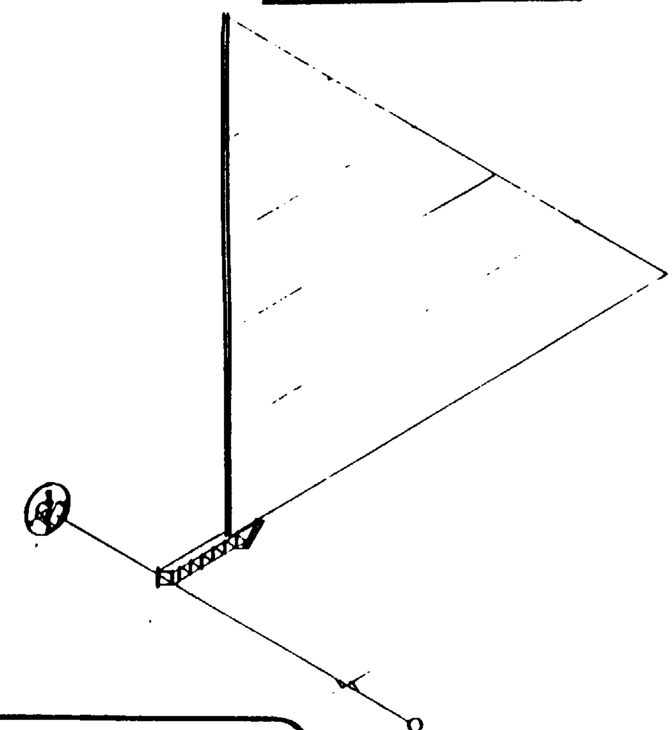
- Spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~4 km)
- Single thruster pod
- 8 solar arrays - "fan" deployment
- Thrusters and arrays contiguous
- 50% truss savings over option 2A

g_a Solar Electric Propulsion Vehicle
Option 2C

Orthogonal View



Isometric View



- Spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~4 km)
- Single thruster pod
- Single solar array
- Thrusters and arrays contiguous
- Deployable truss - 35% less than option 2A

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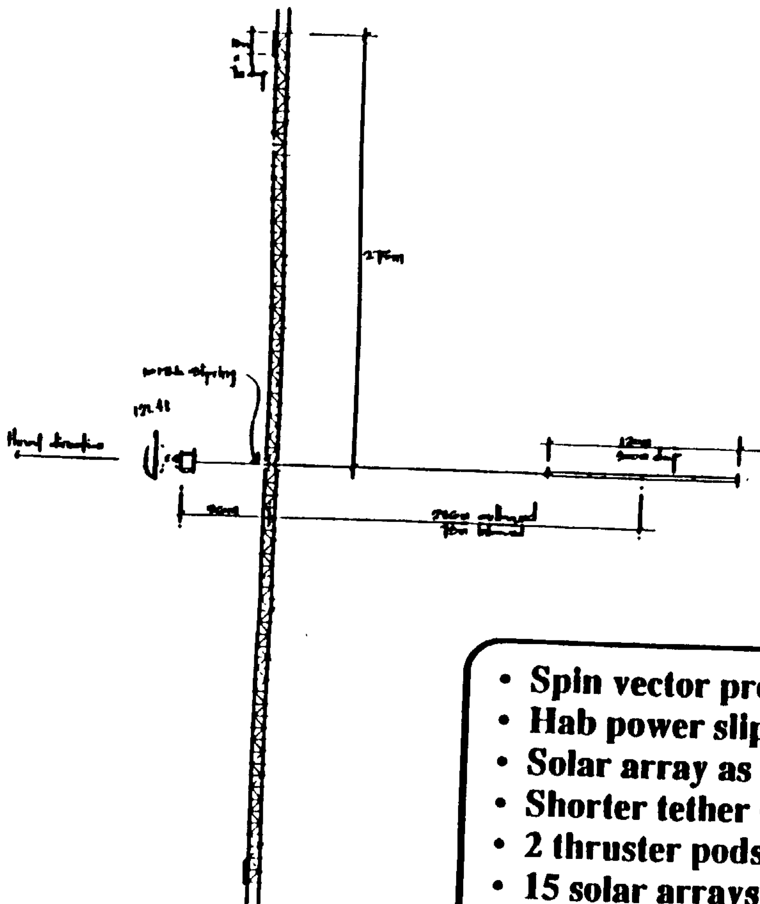
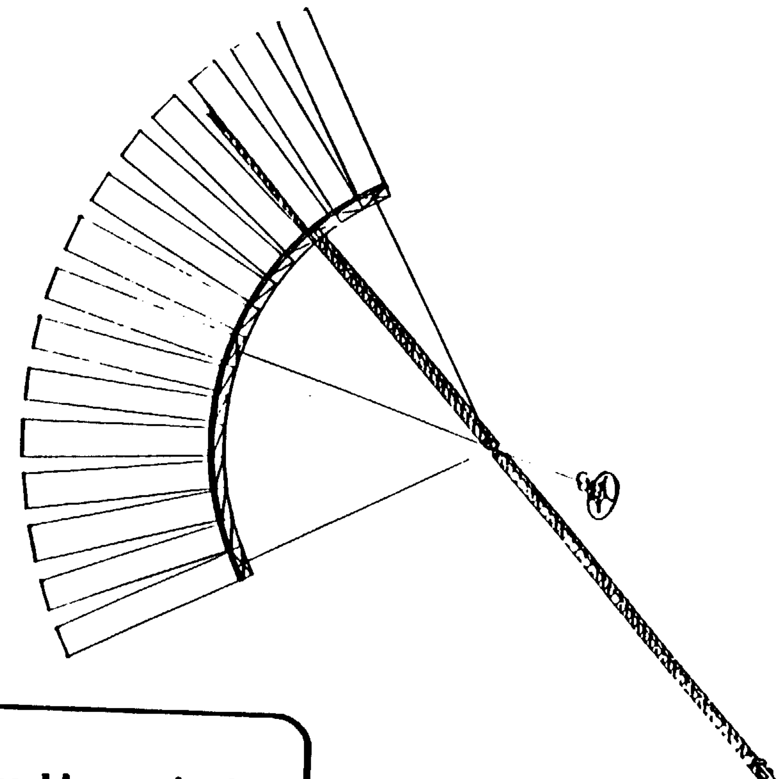
g_a Solar Electric Propulsion Vehicle Option 3

Orthogonal View



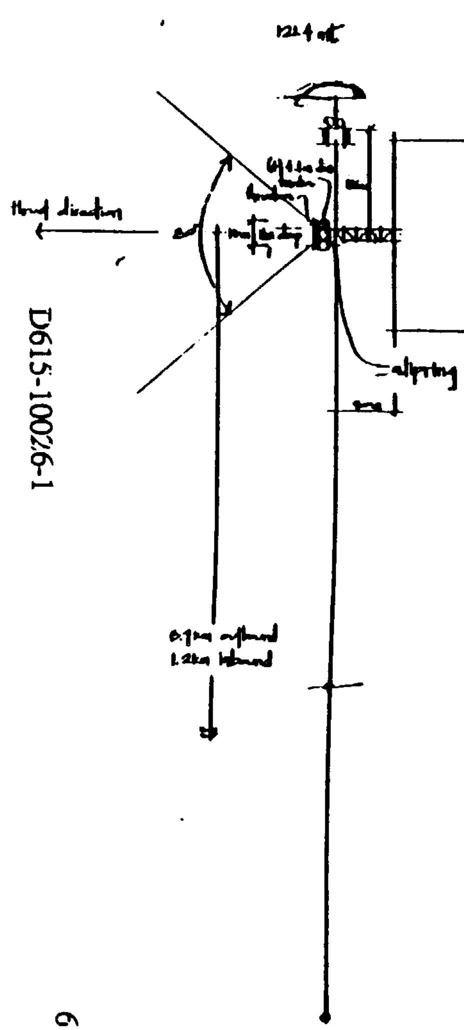
Isometric View



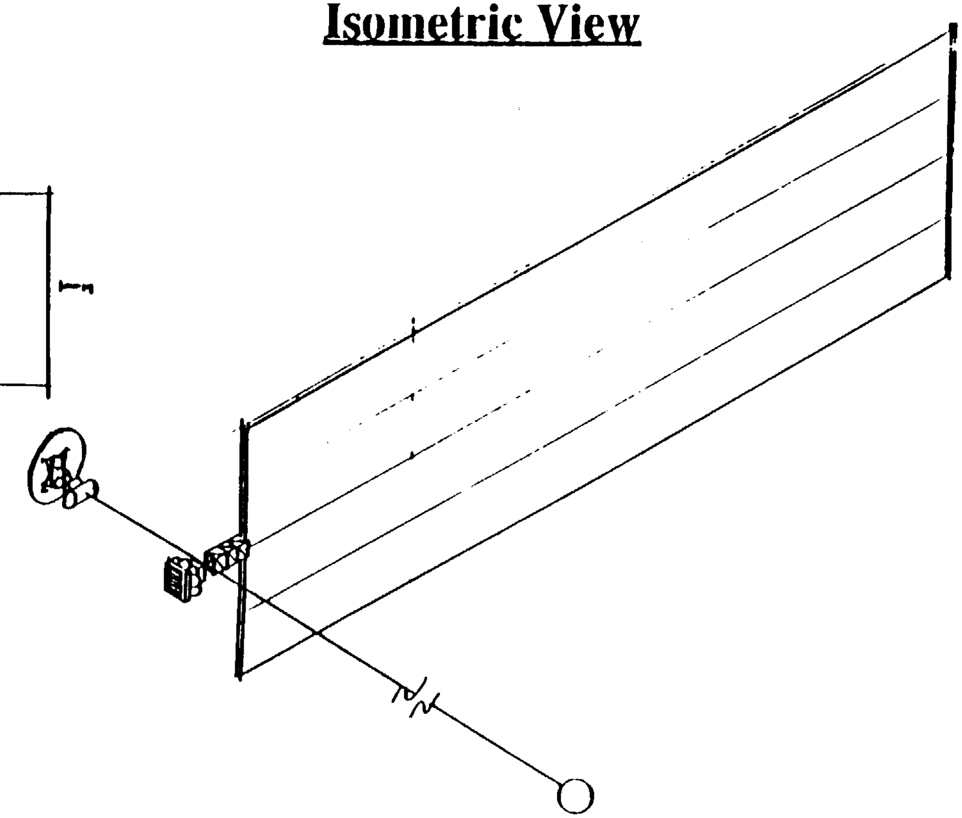
- Spin vector precession
- Hab power slipring assembly required
- Solar array as countermass
- Shorter tether (~250 m)
- 2 thruster pods
- 15 solar arrays - "radial" deployment
- Slipring separating thrusters and arrays
- 550 m of truss

g_a Solar Electric Propulsion Vehicle
Option 4

Orthogonal View



Isometric View



- Spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~4 km)
- Single thruster pod
- Single solar array
- Slipring separating thrusters and

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g_a Solar Electric Propulsion Vehicle
Option 5

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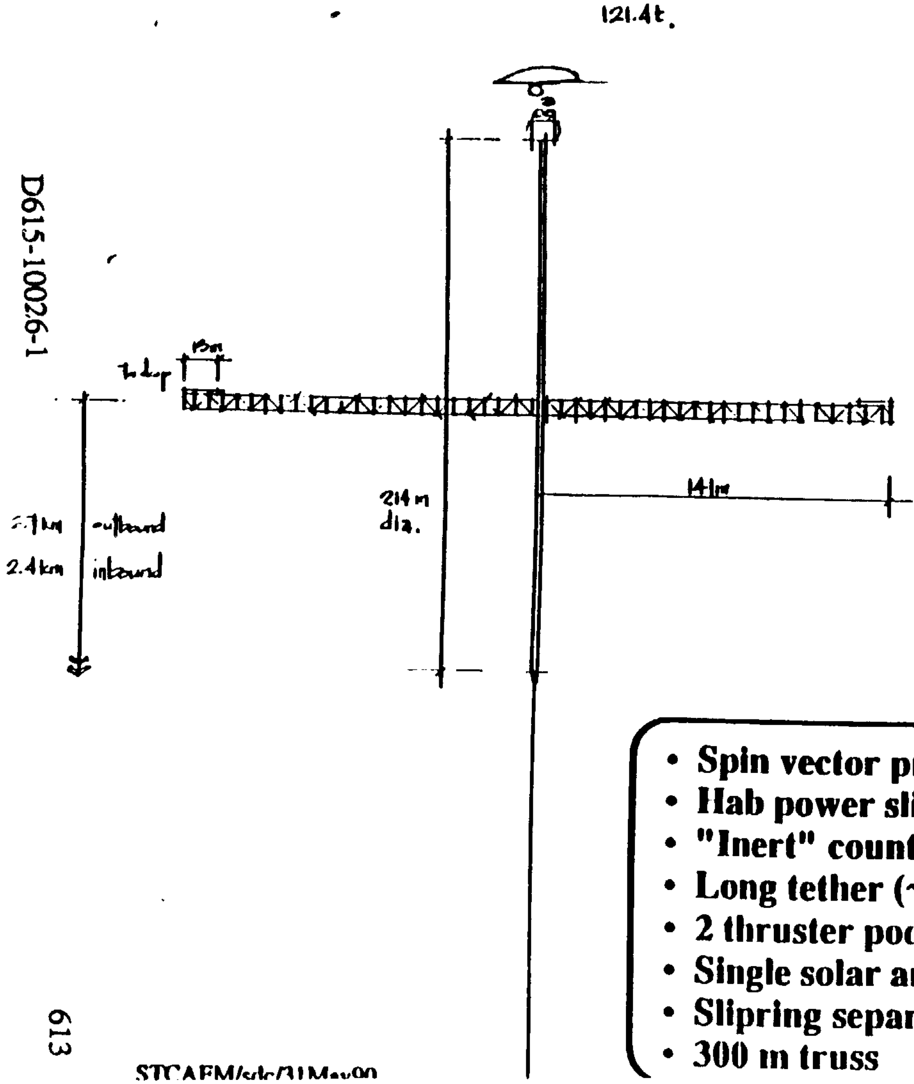
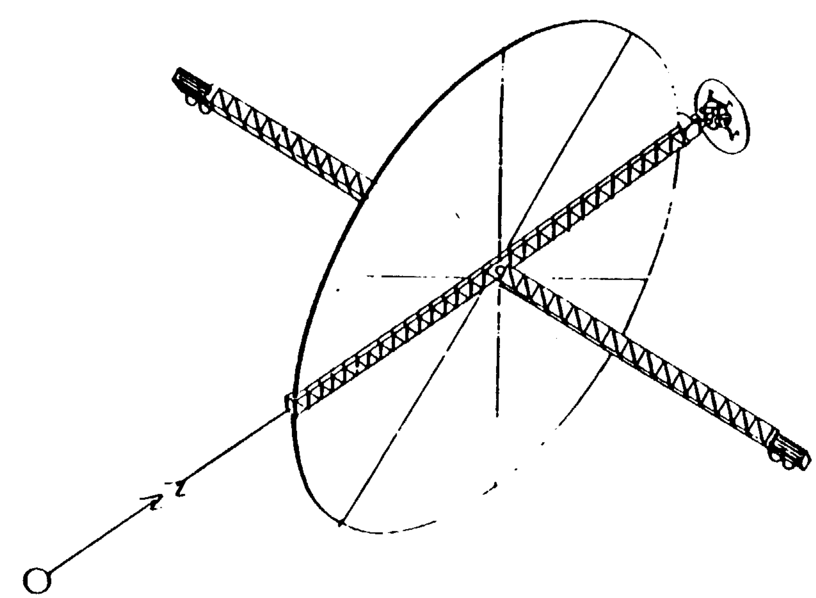
Orthogonal View



Isometric View



- Spin vector precession
- Hab power slipring assembly required
- "Inert" countermass required
- Long tether (~6.6 km)
- 2 thruster pods
- Single solar array - circular configuration
- Slipring separating thrusters and array
- 300 m truss

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Eccentric Thrusting for g_a Electric Vehicles

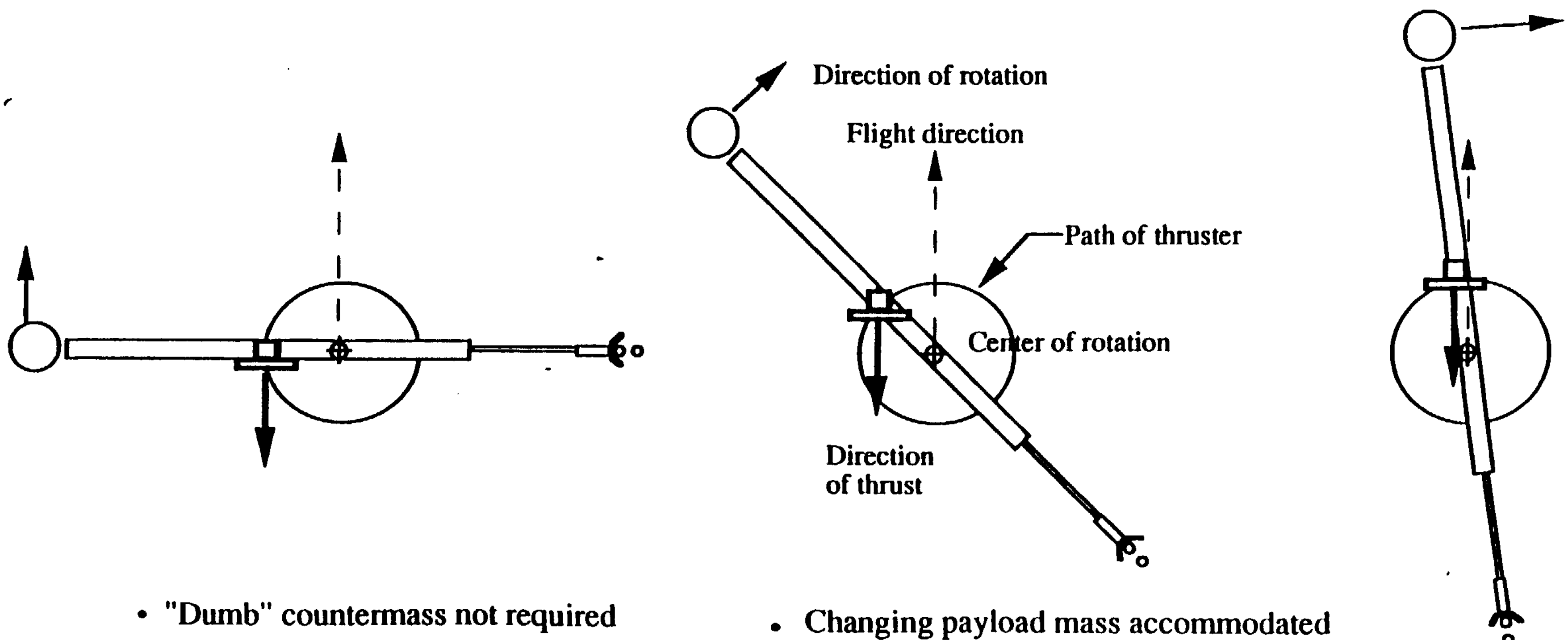
Through studying the g_a SEP problem (in which there is no obvious countermass for the swinging habitation system), a new concept for g_a low-thrust vehicles was developed, called the *eccentric rotator*. It works equally well for the NEP system, and has been chosen as the baseline approach. The vehicle is comprised of rigid structure, with no deployable components. It rotates about its center of mass, which shifts as payload is dropped during the mission. Dual engine assemblies, located on outriggers to prevent ion plume impingement, are despun. Although their thrust vector remains pointed in the same direction, the *source of the thrust (the engine assemblies themselves), orbits the vehicle mass center also*. For low-thrust systems, this is not a problem, because the vehicle's rotational inertia is much greater than can be practically affected by disturbances with a period equal to the gravity-inducing rotation. The moment caused by misalignment of the thrust vector and the CM is cancelled with each 2π rotation. The proper gravity level (1g is baselined) is maintained in the habitation system by adjusting the spin rate.

In a properly balanced vehicle design, the CM remains in the vicinity of the outrigger axis even though it shifts, so rotation-induced moments on the outrigger are always kept small compared to the impulse loading expected from normal assembly, maintenance, attitude-control and berthing maneuvers. As for all rotating vehicles, the structure's stiffness properties must be designed to suppress vibration modes close to the approximately 4 rpm forcing function frequency.

Eccentric Thrusting for Artificial Gravity Electric Vehicles

Rationale: Thruster pods and either solar arrays or radiators can be used as countermass

Conclusion: Continuous low thrust (10^{-4} g) produces an average thrust vector through both the center of mass and rotation



- "Dumb" countermass not required
- Direction of thrust remains constant
- Habitation radius can be fixed

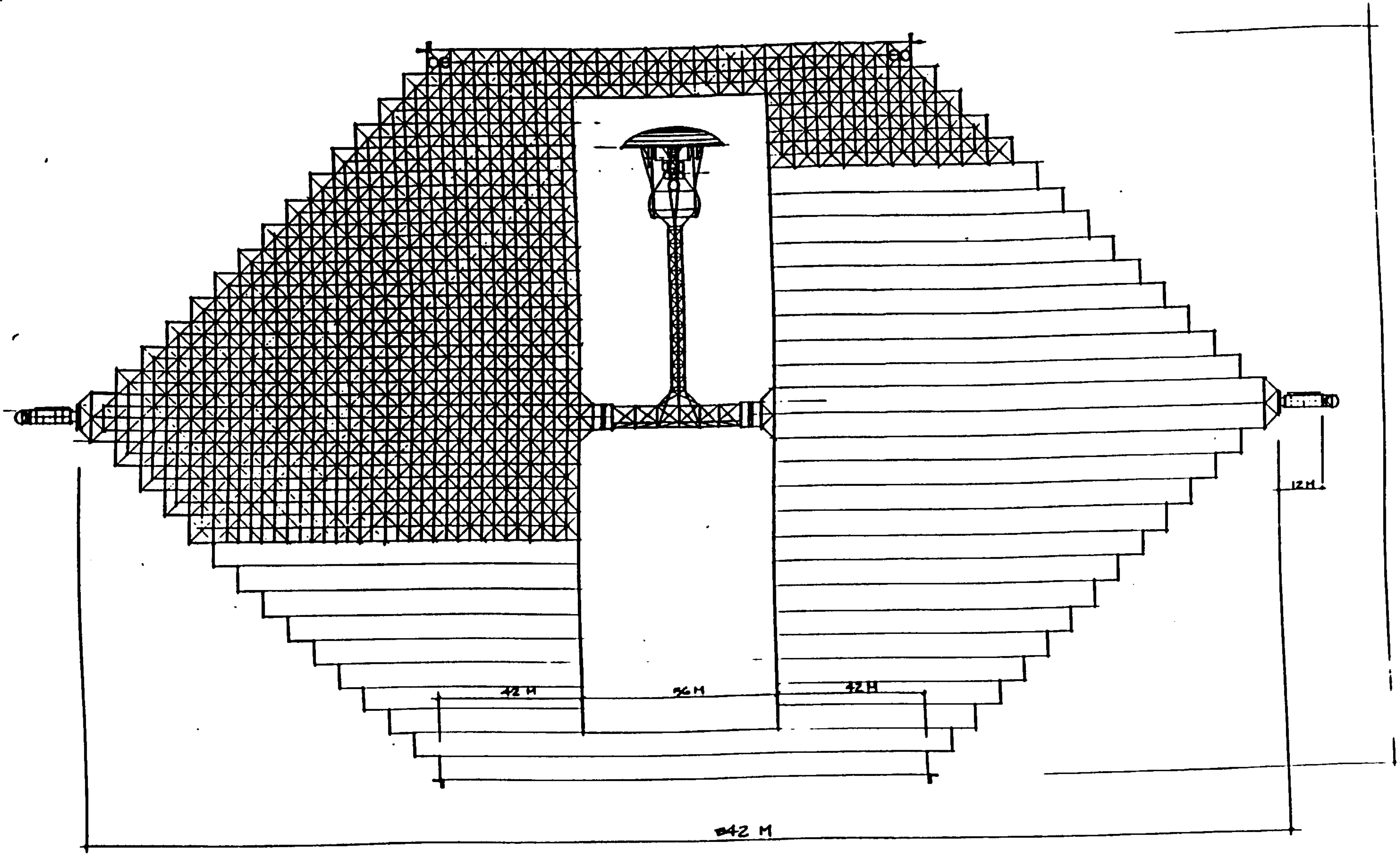
- Changing payload mass accommodated through changing rotation rate
- Minimizes payload structure

g_a SEP Eccentric Rotator Requirement

Shown is the artificial gravity version for the solar electric vehicle. This vehicle uses the eccentric rotating thruster concept, which allows the solar array and truss structure to act as a countermass for the habitat / payload, which revolves around the center of gravity of the entire vehicle at a distance of 56 meters and 4 rpm.

Cyclic loading of the tetrahedral truss structure will have an impact on the total mass of the vehicle, however, this impact has not as yet been determined.

Artificial Gravity SEP



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Nuclear Electric Propulsion Vehicle Artificial Gravity Configuration

The nuclear electric vehicle (NEP) artificial gravity (g_a) concept presents complications not present in the NTR and CAB/CAP concepts. For full-fledged g_a conditions, EP vehicles pose the problem of spinning while thrusting. [An alternative, operational solution may be to fly μg for most of the trajectory, spinning only during the midflight coast intervals (25 to 60 days) and upon arrival at Mars. For STCAEM purposes, however, it is essential to pursue the outcome of a vehicle required to provide artificial gravity for the entire flight.] Because the thrust vector must average tangential to the flight path, the fundamental configuration trade-off is between rotating, high-power transfer assemblies (for the spin vector normal to the ecliptic) and spin-vector precession (for any other orientation).

Of the many possible configuration options identified by STCAEM, the one was chosen that is similar both to the μg NEP and to the SEP g_a concept. This configuration concept, called an *eccentric rotator*, avoids tethers, complex extendible booms or deployable trusses. All components are rigid and the design is simple.

The fundamental concept is that the spine of the μg NEP configuration is intersected orthogonally by a lightweight, symmetrical engine outrigger. The ion engine assembly is split between the two ends of this outrigger, and these are despun from the rest of the vehicle so as to remain properly oriented for thrusting throughout the flight. No deployment mechanism is required to change the habitat system separation when the MEV mass is lost. Instead, the rotation rate is adjusted to provide 1g in the center of the long-duration habitat, *according to the habitat's actual separation from the current vehicle mass center*, which shifts after MEV operations. Thus the mass center is not necessarily axially aligned with the engine outrigger, although it always remains at the zenith relative to the habitat floors. When the mass center is not along the outrigger axis, the outrigger also orbits the mass center. The engine assemblies therefore trace out circles as they thrust, although the thrust vector orientation remains fixed. For low-thrust systems in particular, this is expected to cause no problems. The reactor/power assembly along with the primary radiators are used as the countermass to the crew systems and the secondary radiators.

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g_a Nuclear Electric Propulsion Vehicle

The next 2 charts show the options being considered for the NEP artificial gravity configuration. The NEP is in the preliminary stage of trading configurations, and one concept has not yet been chosen.

ga NEP Concept Development

Configuration Trades Performed

- *Ecliptic-normal spin vector* vs. cross-product engine
- Shield size vs. reactor separation (*50° half-angle*)
- *Straight* vs. swept-back engine outriggers
- Structure deployment mechanisms (*telescoping truss*)
- Structure deployment vs. *eccentric rotator*
- *Roll-ring* vs. slip-ring vs. liquid-metal bearing

Concept Updates Implemented

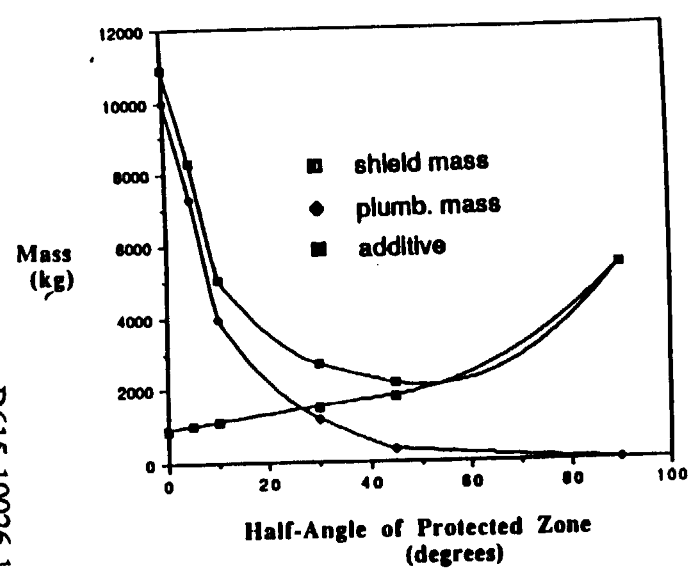
- Counter-rotating turbomachinery function combined into single housing
- Radiator heat-pipe length doubled to 30 m
- Heat pipe axes oriented normal to gravity vector
- Double shields collapsed into one
- "One-dimensional" shielded vehicle aspect
- Shield shaped to reflect vehicle geometry
- Configuration pre-adapted for multiple MEVs

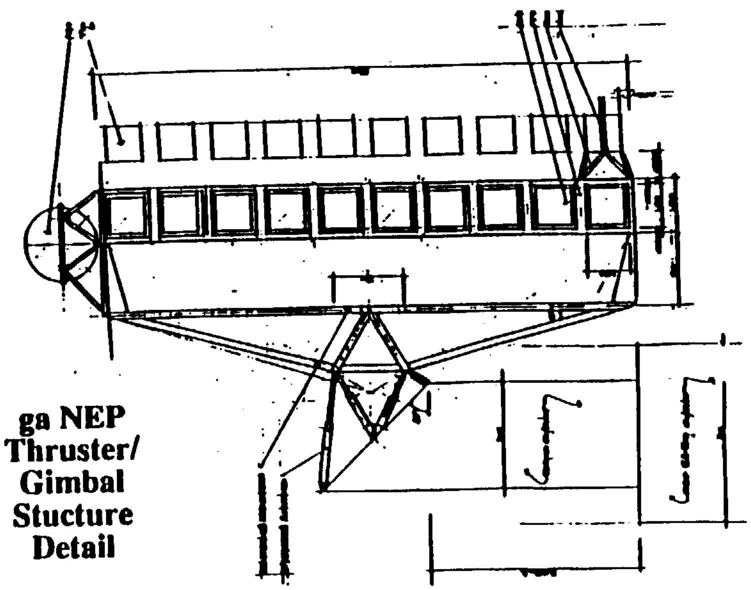
g_a NEP Concept Development Details

Shown are a few of the configuration detail analyses performed to resolve integration issues for the g_a NEP vehicle concept, as well as the result of a fundamental trade done to determine the best combination of vehicle length and shielding angle for both the μg and g_a configurations.

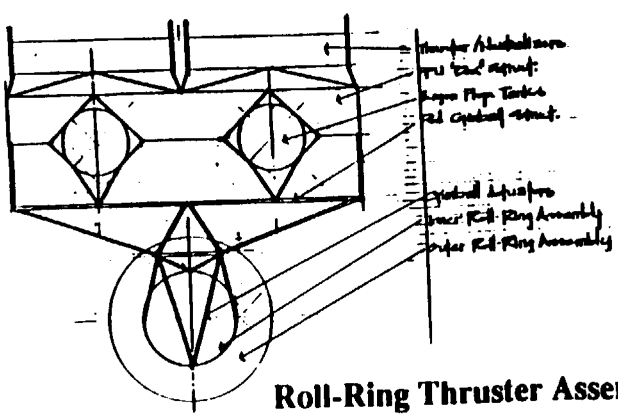
Artificial Gravity Concept Development

Shielding Angle Optimization Trade

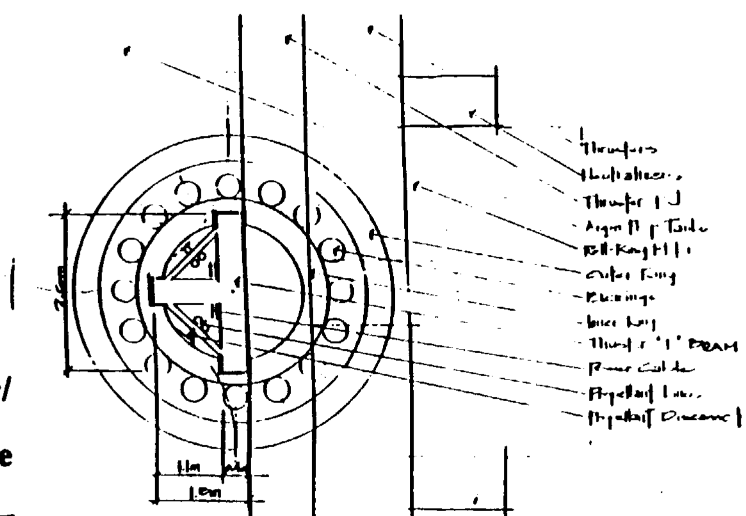




ga NEP Thruster/ Gimbal Stucture Detail



Roll-Ring Thruster Assembly



NEP Thruster/ Gimbal Structure Detail Top View

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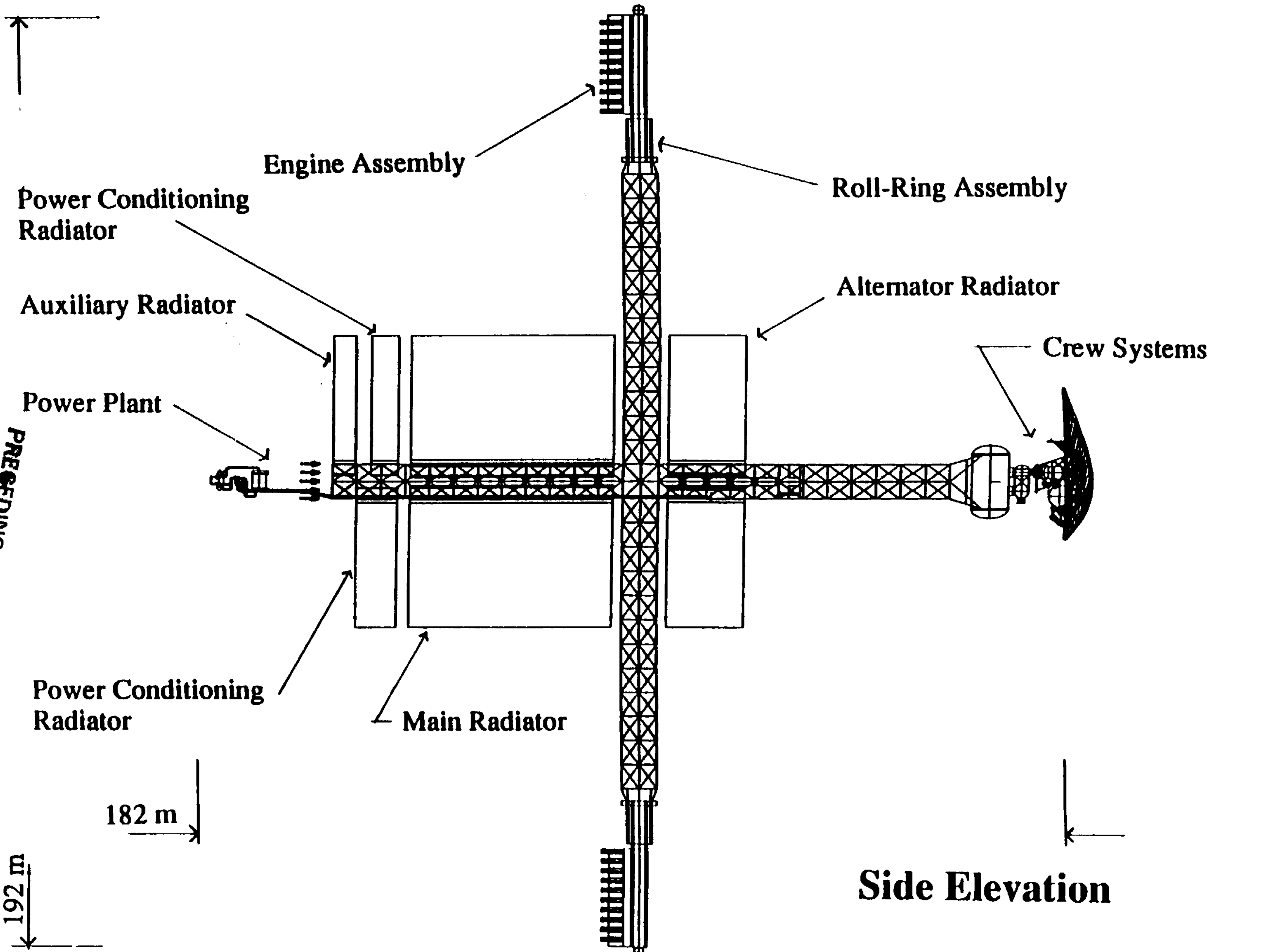
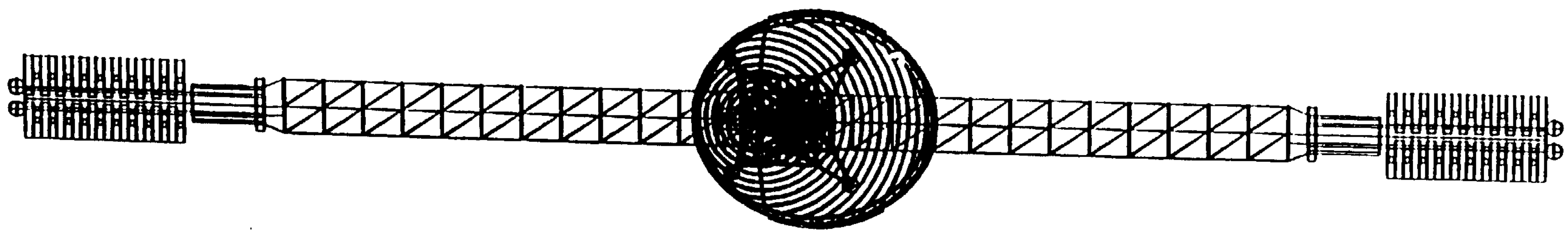
g_a NEP Configuration

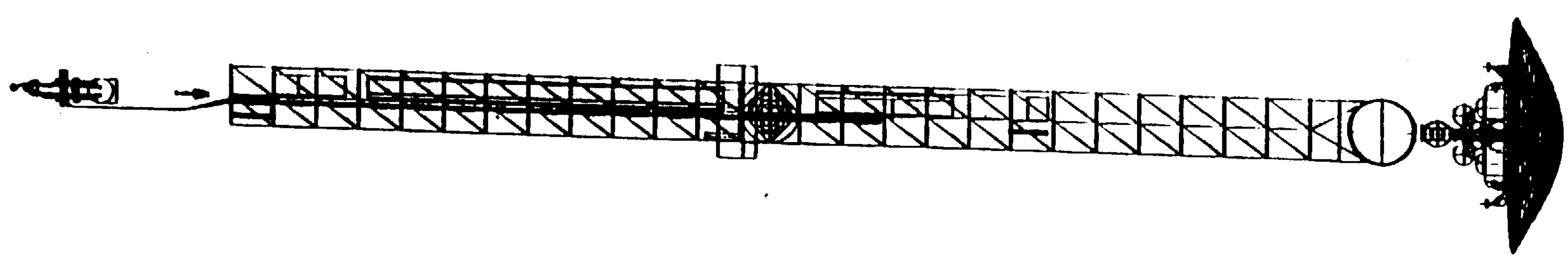


Side Elevation

g_a NEP Configuration

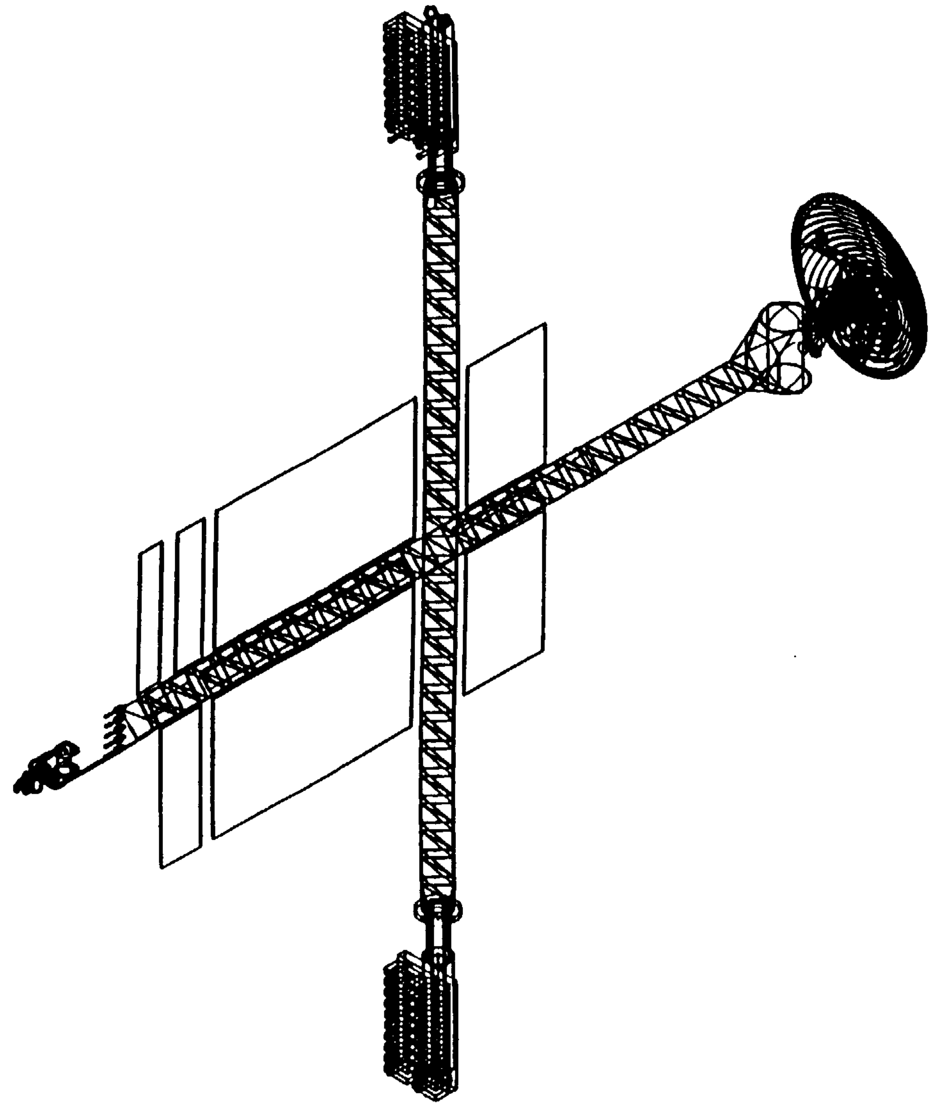


Front Elevation



Top Elevation

g_a NEP Configuration



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Artificial Gravity NEP Mass Statement

Mass in metric tonnes

Payload

Descent Aerobrake	7.0
MEV Descent Stage	18.7
MEV Ascent Stage	22.5
Surface Equipment	25.0
Transit Hab Module (for 4 crew)	44.3
	117.5

Propulsion

Reactor 1	7.4
Reactor 2	7.4
Shield	8.6
Primary Heat Transport System	20.1
Auxiliary Cooling Subsystem	2.2
Boiler	21.6
Turboalternators	16.3
Alternator Radiator	2.6
Turbopumps	.4
Rotary Fluid Management Device	3.1
Main Cycle Radiator	10.6
Main Cycle Condenser	1.3
Main Cycle Plumbing	5.0
Auxiliary Cycle Radiator	3.3
Auxiliary Cycle Condenser	1.3
Auxiliary Cycle Plumbing	6.0
Power Conditioning Radiator	1.1
Plumbing Insulation	4.1
Engine Assembly	23.5
Power Management & Distribution	68.0
	211.1

Structure

5 Meter Bay Graphite-Epoxy Truss	8.1
Pressurized Berthing Adaptor	6.6
	14.7

Utilities

Communications	.6
Attitude Control	5.7
Avionics	2.5
Houskeeping Power Distribution	.5
PV/RFC Power Subsystem	2.3
40 Mwe Roll Ring	12.0
Robotics	7.2
	30.8

Tanks	3.4
Feed Lines	0.1
Propellant	171.7

Total	**549.3**
15% growth	**38.5**
IMLEO	**587.8 t**
Resupply Mass	**355.5 t**

Trip Time = 520 days, alpha = 7.4 kg/kW

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MTV/MEV Mission Scenarios

This section shows several items that must be considered in mission scenarios to define unified requirements for a complete mission; that is coordinating the operations of the MTV and MEV, particularly those areas that are of concern to LEVEL II .

The difficulty of a mission is defined by several factors. This includes the time frame the mission is to be conducted; the physical positioning of the planets driving the degree of difficulty to reach and return from Mars. The total mission ΔV (change of velocity) is a good indication of the physical "cost" of the mission. Shown here are full mission contours for two conjunction class missions (2010 and 2013) for the given staytimes at Mars . The x-axis is in Julian date 245XXXX. The y-axis is in trip time, inbound or outbound, in days. The ΔV minimums show clearly on the contours.

Arrival at Mars must also be taken into account as when the S-vector (return trajectory vector) may be out of plane and the true anomaly is not close to periapsis to impose a sizable ΔV penalty to acquire the trajectory home. These will be functions of the orbit period and inclination. The information given is for the 2010 Conjunction mission shown in the preceding chart. To further the evaluation the position of the periapsis with respect to the surface (defines access to the surface) and the lighting angle at periapsis (daylight or night landing) must be identified. For this case a capture period of 10 hrs. gives a minimum departure ΔV of approximately 1.2 km/sec for a 30 degree inclination orbit, an S-vector out of plane by >2 degrees, a true anomaly off of periapsis by ~ 10 degrees (minimum) , a periapsis latitude of ~30 degrees north (out of the expected permafrost region on the planet), with a periapsis lighting angle of + 40 degrees (daylight). This means that the capture is in daylight , an abort possibility exists that is easily accessible, and the craft is positioned over areas of interest on the surface.

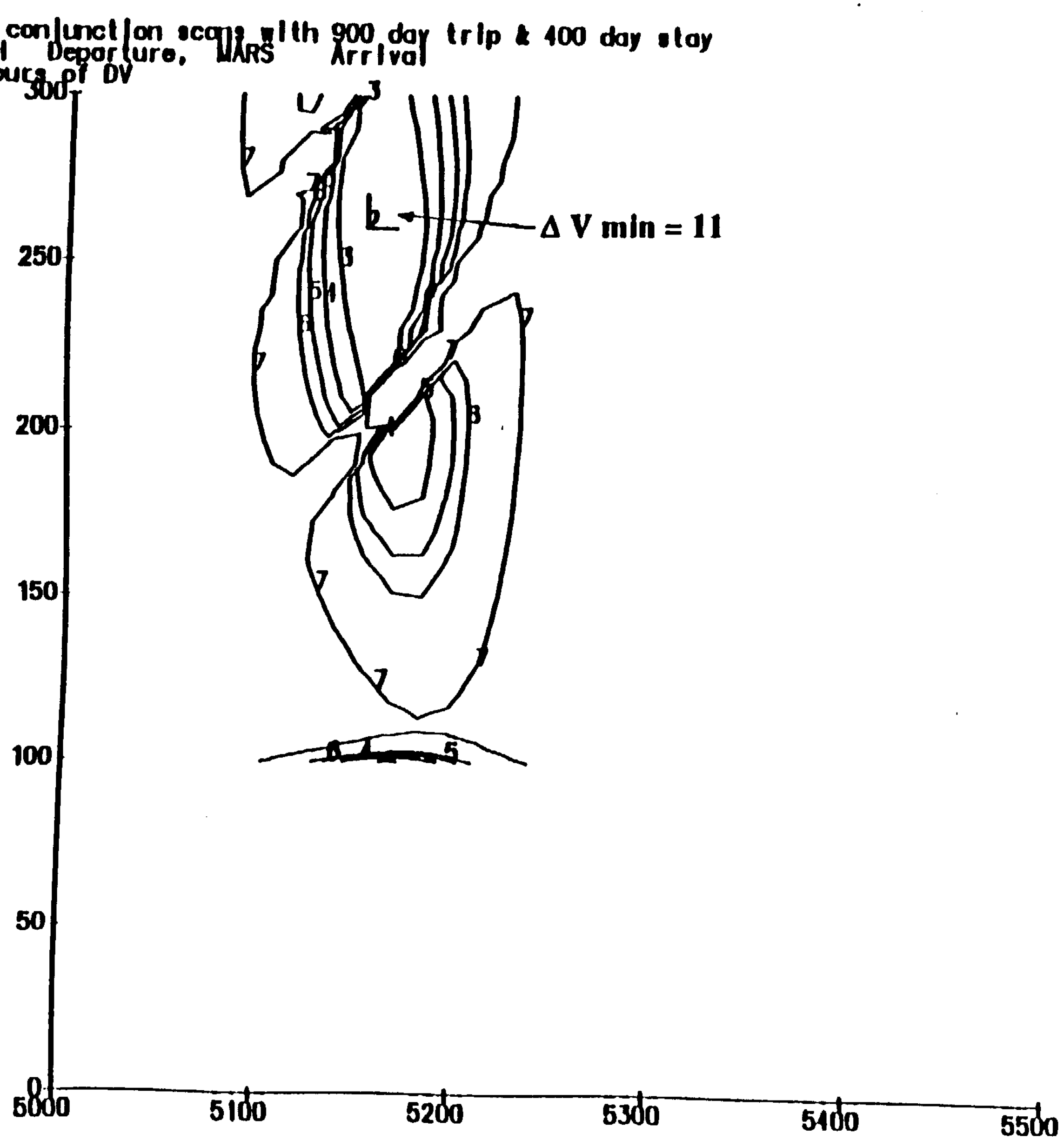
Once on the surface the propellant choice made for the lander and the length of surface stay, dictate the weight of the lander. In the case shown, a minilander capable of reaching a 250 km orbit was used to trade the weight of the vehicle for the type of fuel used. Both storable and cryogenic fuels were employed and the cryogenic systems accounted for the boiloff and the extra decent propellant needed to land the boiloff propellant. Even at 600 day staytimes with boiloff considered, the cryogenic systems trade favorably with the storables. However, since the atmosphere of Mars is present, the cryogenic systems must be vacuum jacketed and have abort procedures in case of an atmospheric leak in the jacket, which would permit CO_2 ice to form around the tank.

The values for the boiloff were generated by using an in house Boeing program to calculate the boiloff, and boiloff rate for various sizes, with or without MLI, vapor cooled shields,and pera-to-ortho conversion of hydrogen. This same program was used to predict tank properties for an External Tank (ET, current Shuttle) sized volume and the resultant tank masses and tank fractions for use in weight estimations for cryogenic propellant systems on all vehicles.

2010 Conjunction minimum round trip ΔV is approximately 11 km/sec with an outbound/inbound trip time of slightly greater than 250 days.

2010 - 400 Day Stay
900 Days Round Trip

Contour	ΔV
1	10
2	11
3	12
4	13
5	14
6	15
7	20

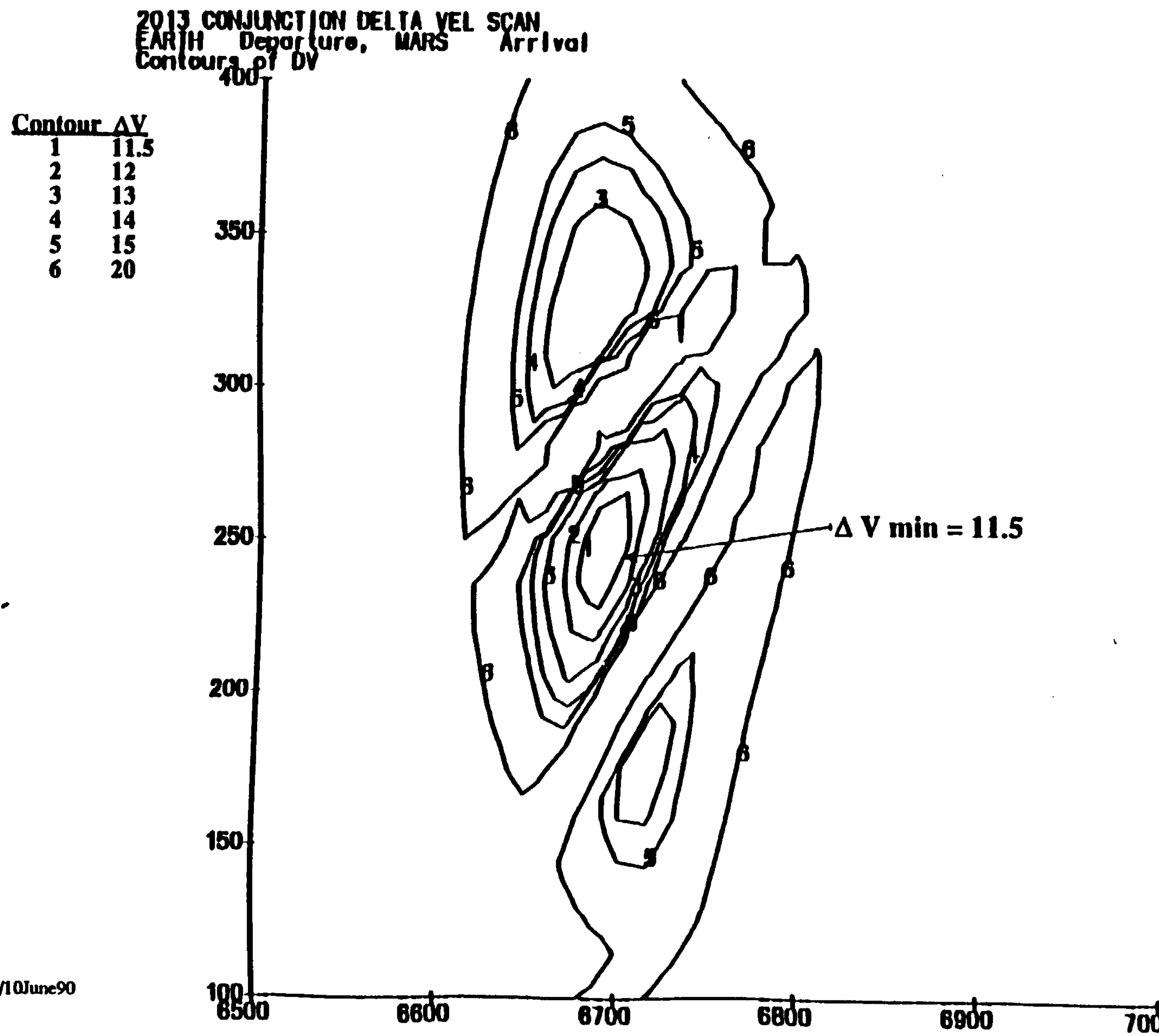


STCAEM/grw/10June90

2013 - 400 Day Stay -- 900 Days Round Trip

2013 Conjunction minimum round trip ΔV is approximately 11.5 km/sec with an outbound/inbound trip time of approximately 240 days.

2013 - 400 Day Stay
900 Days Round Trip



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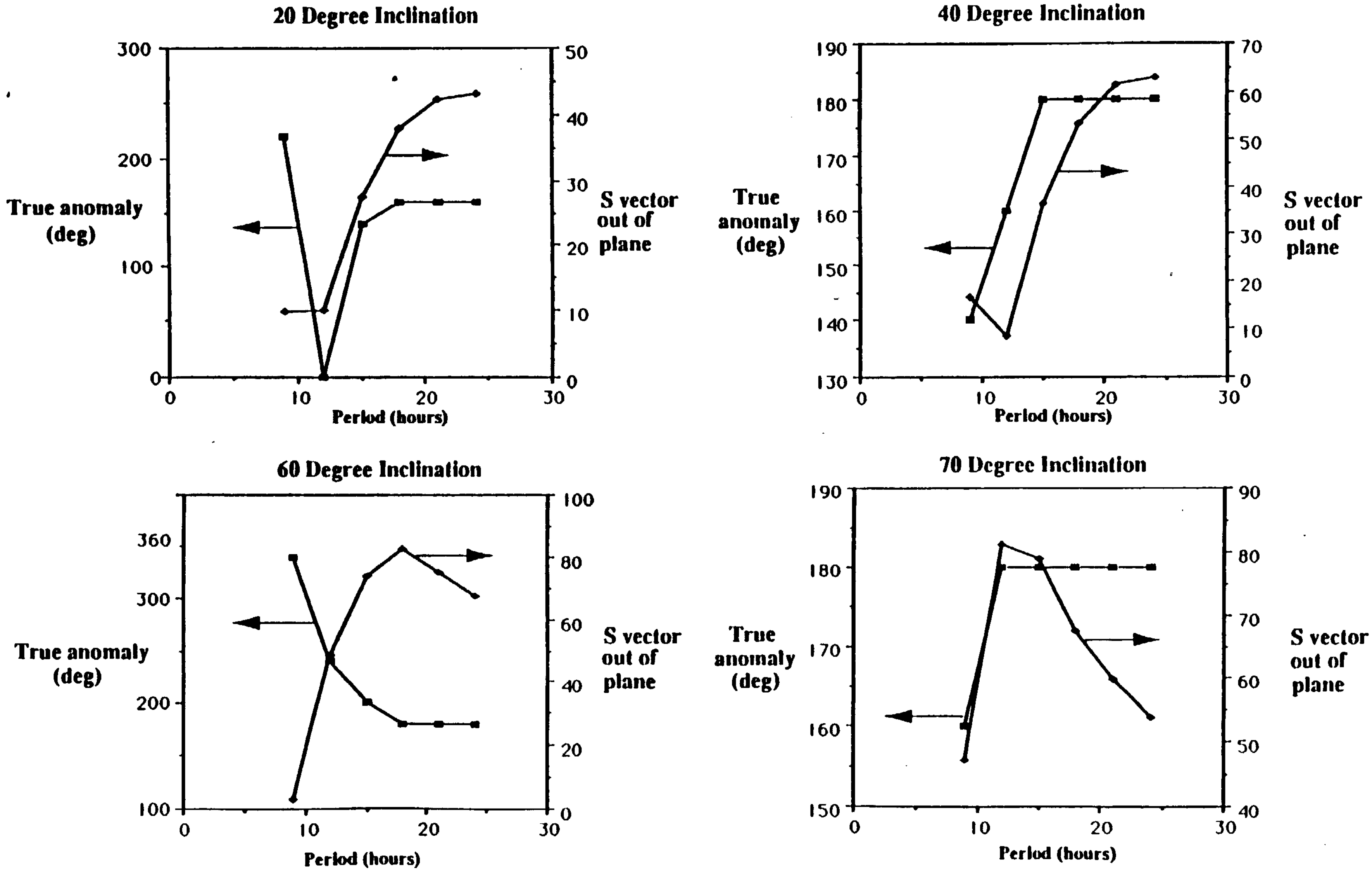
2010 Conjunction
S Vector and True Anomaly vs Parking Orbit Period

Minimum departure delta V occurs when the S vector is in plane and departure true anomaly is close to periapsis.

2010 Conjunction
S Vector and True Anomaly vs Parking Orbit Period



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2010 Conjunction Mission
Orbital Parameters vs Parking Orbit Period

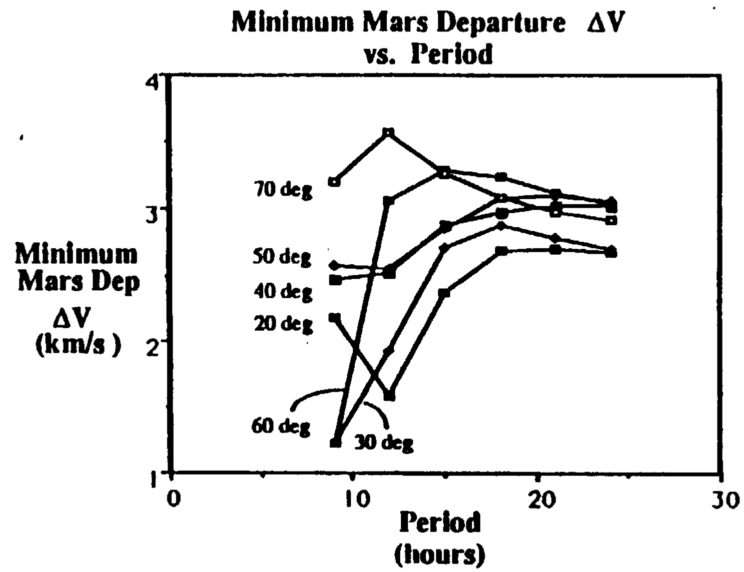
Minimum Mars departure ΔV of approximately 1.2 km/sec occurs for parking orbits with inclinations of 30° and 60° and period of 9 hours each.

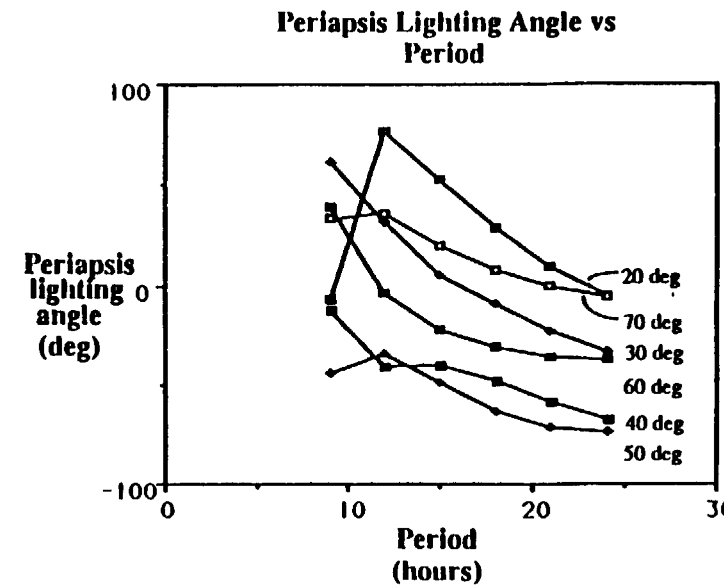
Periapsis lighting angle is adequate for the parking orbits with inclinations of 30° and 60° and period of 9 hours each.

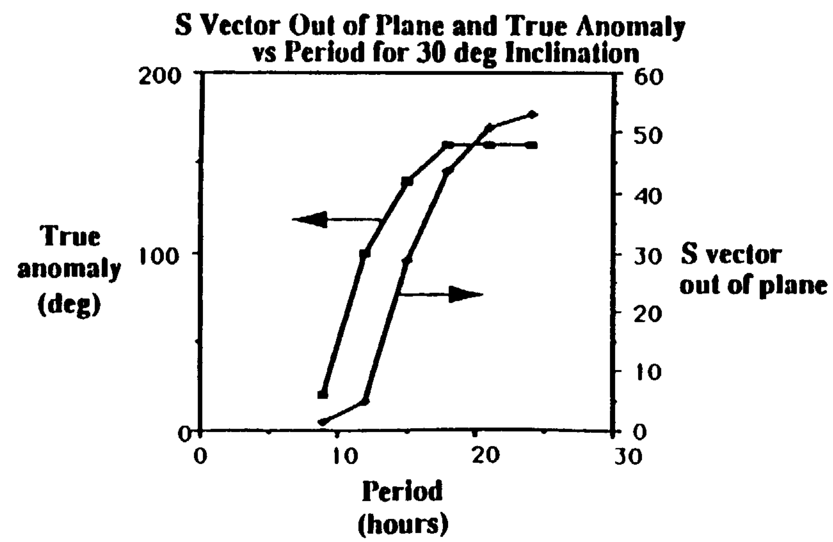
Minimum S vector out-of-plane occurs when the true anomaly at departure is closest to periapsis and the orbital period is 9 hours.

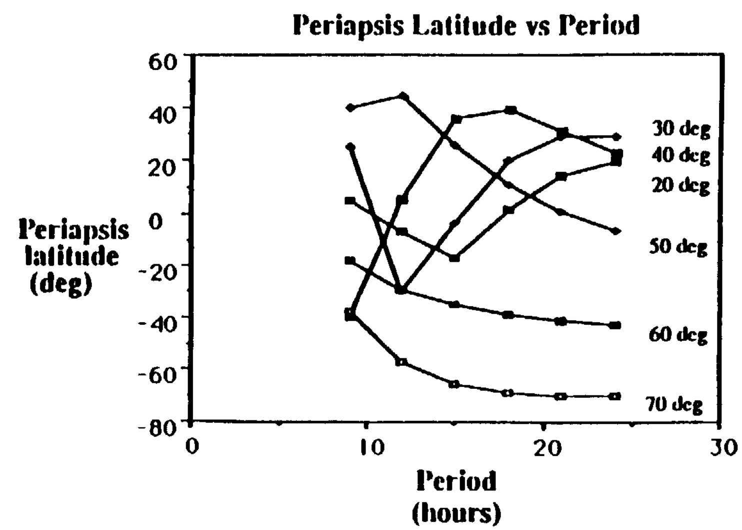
Periapsis latitude for a 30°, 9 hour parking orbit provides a landing coverage for landing sites between 38° to 50° north latitude; periapsis latitude for 60°, 9 hour parking orbit provides landing site access to landing sites between 5° south to greater than 20° north latitude.

2010 Conjunction
Orbital Parameters vs Parking Orbit Period









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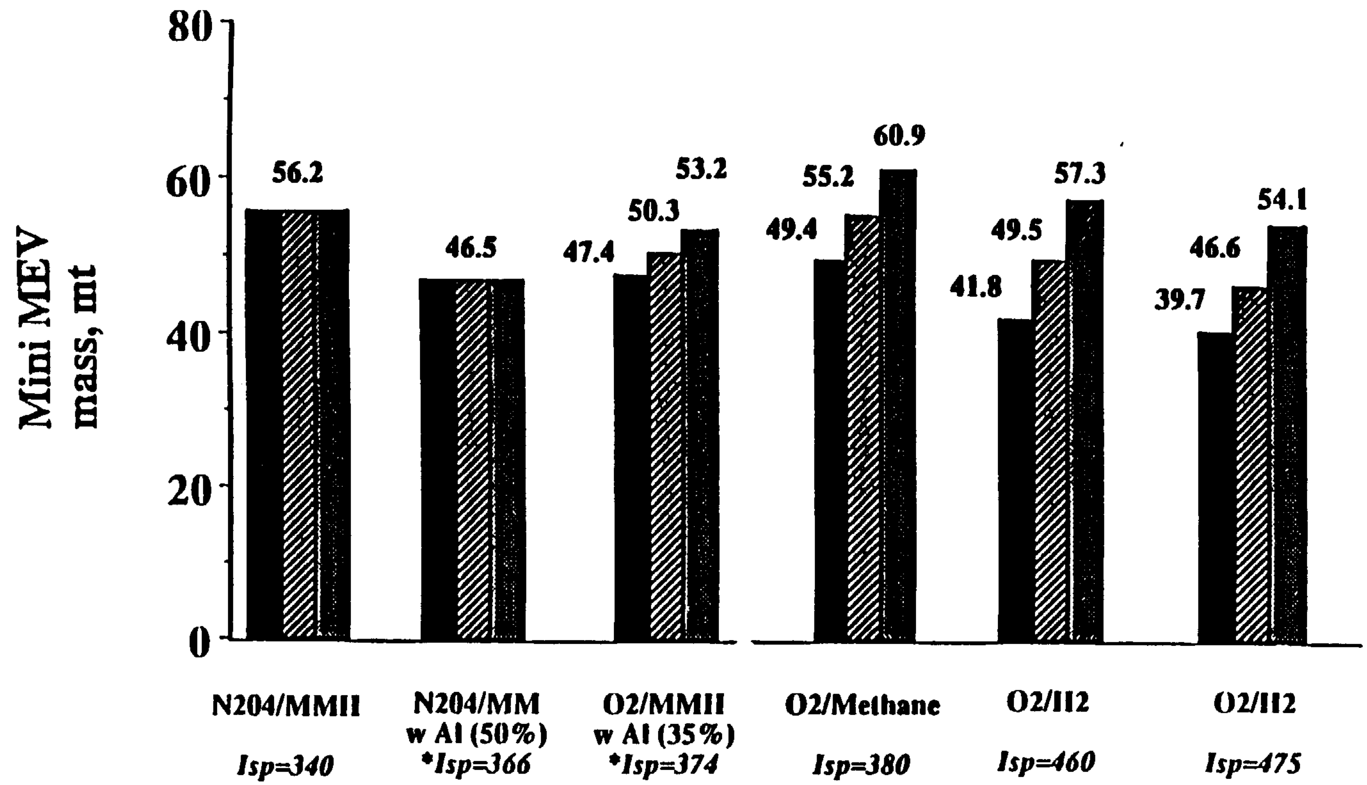
Mars mini lander propellant trade

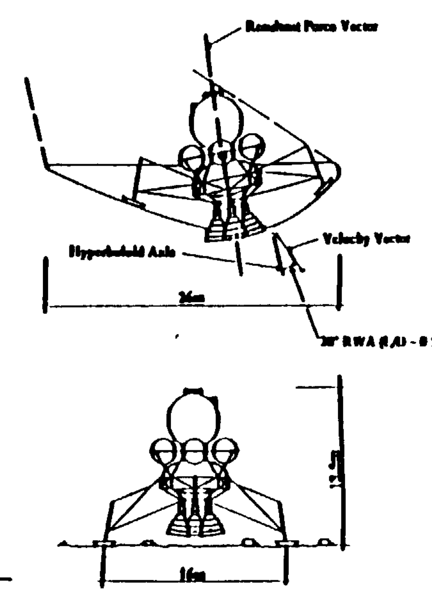
The Boeing mini MEV craft was designed to be one half the weight of the reference MEV that carried a 25 t cargo load to the surface, in order that two landing missions might be done for the same payload cost as the reference. The minilander as designed carries a crew of 3 instead of 4, had a shorter stay time capability, 6 to 8 days vs 30 days, and had less capability to carry science payload to the surface. However the ability to land at two separate sites per mission and/or to have a backup lander made the idea attractive. This trade evaluated the use of 5 different propellant combinations for the single stage vehicle. Three surface stay times were used to determine the effect of propellant boiloff on lander mass as well.

<u>Results:</u> The storable propellants, though lower in performance, have no boiloff penalty as the cryogenic propellants do, and are shown in the first two cases. The second of these two has Aluminum added to the MMH in order to raise the Isp, in this instance 26 sec rise to 366 sec. This figure was obtained from Bryan Palaszewski of NASA/LERC and is based on the CEC propellant computer code with an efficiency factor of 93% assessed. Three cryogenic combinations follow, with the reference cryogenic mini MEV total mass = 39.7 t for a 10 day stay with an Isp set at 475 sec. The Boeing <u>Crystore</u> computer model was used to calculate boiloff components for the 300 day and 600 day stay times.

Mars Minilander Propellant Trade

Revision 2 5/18/90





- *Single stage Desc/Asc vehicle, crew of 3*
- *Cryo vehicles: 4 LO2 tanks; storable veh's: 1 N2O4 tank*
- *Crew occupancy time in Minilander = 6 to 8 days*
- *Surface stay provisions assumed already implaced on surface for stay times greater than 10 days*

* Bryan Palaszewski NASA/LeRC 12/20/89
/STCAEM/bbd/14June90

synthesis model run #: minilander.dat;135/136/137
Mac chart: minilander propel trade 5/18/90

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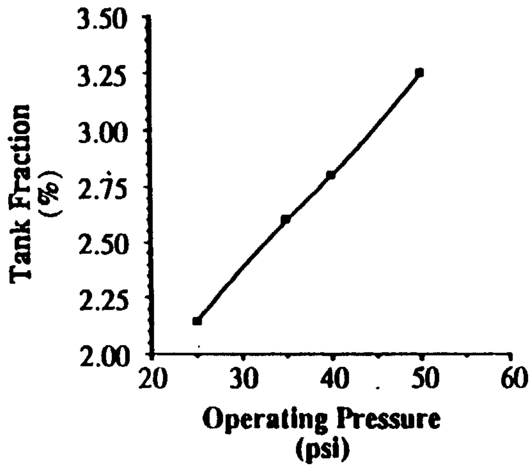
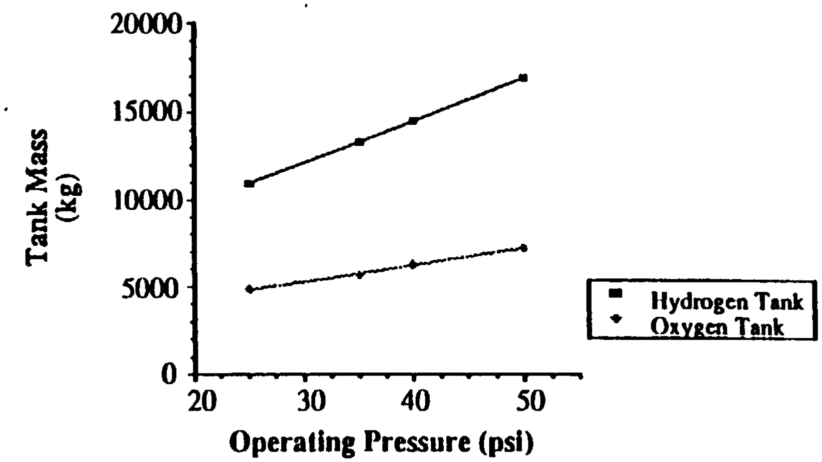
Cryo Boiloff Code Tank Properties Prediction for ET Sized Tanks

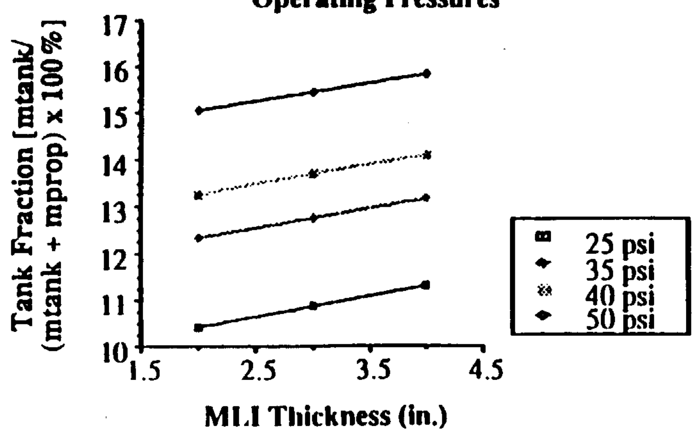
Overall Tank Fraction vs. Design Pressure for ET Sized Tank-sets



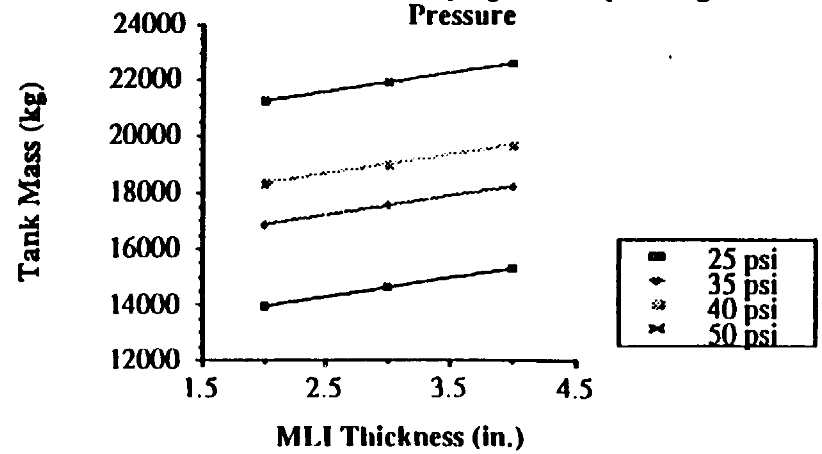
Tank Mass vs. Design Pressure for ET Sized Tank



NTR Hydrogen Tank Fraction vs. MLI Thickness For Varying Tank Operating Pressures



NTR Hydrogen Tank Mass vs. MLI Thickness For Varying Tank Operating Pressure



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Comparison of Predicted Tank Masses & Fractions With the External Tank

- External Tank Data -

Total Dry Mass = 35,425 kg
Propellant Load = 719,112 kg
  LH2 Load = 102,618 kg
  LOX Load = 616,493 kg
LH2 Tank Mass = 14,402 kg
LOX Tank Mass = 5695 kg
Tank Max. Operating Press:
  LH2 - 34 psi
  LOX - 22 psi

Overall Tankage Fraction = 4.7 %

LH2 Tank Fraction = 12.3 %
LOX Tank Fraction = 0.91 %

LH2/LOX Overall Tank Fraction = 2.72 %

- Boiloff Code Predictions -

Assumptions:
  MLI Thickness = 2"
  Diameter = 4.2 m
  Tank Shape - Cylindrical tank with √2 ellipsoidal endcones
  Ullage = 5 %
  Propellant Mass = ET Propellant Loads
  Mass includes vapor cooled shields, supports, and para-to-ortho H2 converter, where appl.

Results:

Tank	Pressure	Tank Mass	Tank Fraction (overall)	
H2	35 psi	13306 kg	11.5 %	(2.6 %)
O2		5730 kg	0.92 %	
H2	40 psi	14521 kg	12.4 %	(2.8 %)
O2		6229 kg	1.0 %	
H2*	50 psi	16951 kg	14.2 %	(2.99 %)
O2**	30 psi	5239 kg	0.84 %	

* NTR value-25 psi x 2 for overpress.
* 2 x overpress. allowance

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Aerobrake Summary

This section is a summary of Aerobrake related information as it pertains to the Boeing work on the STCAEM contract managed by the NASA Marshall Space Flight Center. This summary addresses the aerobrake analyses categorized as geometric configuration for capture and landing, Mars atmosphere knowledge uncertainty impacts on GN&C, design configurations for reducing heating rates and loads, landing flight mechanics for range and crossrange requirements, structural techniques for reducing weight, and integration of technology to meet overall mission goals. The aforementioned categories will be covered in four sections: Aerocapture, Heating, Structure, and Ascent/Descent.

Aerocapture - Critical GN&C related aerocapture issues are line-of-apside control and apoapsis altitude control. Aerocapture analyses results included in this summary show the following:

* Asymmetric roll with a finite rate provides improved line of apsides control.

* A guidance system designed for a low density atmosphere needs to be optimized for other atmospheric conditions.

* Using MarsGram, a one sigma density change results in a large difference in density variation between day and night.

* The guidance system (as related to aerocapture exit conditions) is more affected by large (wavelength > 1000 km) horizontal sine wave density variations.

* A larger vertical wavelength (on the order of 20 km) sine wave density induces a lesser error than a smaller vertical wavelength (on the order of 5 km) sine wave density.

Heating - Mars aerocapture heating analyses results are given for stagnation point heating and for some choices of stream line point heating. Heating analyses results included in this summary indicate the following:

* For the Mars aerocapture MTV, the stagnation point heating rate resulting from averaged lift-down L/D is lower than the heating rate for average lift-up L/D.

* Under similar conditions, the heating loads follow the same trend as the stagnation point heating rate.

* Along the center streamline of the hyperboloid aerobrake the predicted radiative heat transfer rate at Mars using the Park method is approximately two times that using the Tauber-Sutton method.

* The total heating rates at the stagnation point with Park (146 w/sq cm) and Tauber-Sutton (80 w/sq cm) are higher than the near term (1993) radiative material capabilities of approximately 70 w/sq cm.

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* For an averaged L/D = 0.5 the stagnation point heating rate for Mars aerocapture is 146 w/sq cm; Earth aerocapture heating rate is 172 w/sq cm.

* The local Reynolds number along the aft streamline of the 30m body does not exceed 10E6.

Structures - Structural analyses results demonstrate weight savings and strength improvements through advanced composites application and through spar design advantages. Included structural analyses results depict the following:

* Spar and truss configurations were developed for the 30 meter aerobrake concept.

* For the spar configuration and with current technology, the (81 mt payload) weight estimate is 41.5 klb and the MTV (153 mt payload) estimate is 66.3 klb.

* Improved material characteristics (200 ksi vs 105 ksi span strength) reduces configuration weight by greater than 15%.

* Mass savings of 30% may be achieved by improved spar design and advanced materials characteristics.

Structures - (cont.)

* The truss configuration provides a 15% weight savings compared to the spar configuration.

Ascent/Descent - No ascent related information is discussed in this version of the IP&ED; a forthcoming update will contain a discussion of ascent related data.

Descent trajectory analyses results point to L/D requirements related to landing site accessibility issues. Included descent trajectory results include the following:

* For MEV with L/D = 1 and descent inclination of 45 degrees, a displacement in latitude of 30 degrees may be achieved.

* An increase in L/D from 0.5 to 1.0+ extends the range by approximately 50%.

* An aeroflare reduces the ideal delta velocity required for landing by 200 to 300 m/sec (L/D = 1).

(4)

Topics discussed in the analyses include configuration parameters (L/D, M/C_DA) required for capture and landing. The parameters are also utilized to satisfy range and landing requirements of the vehicle. The configuration of the vehicle is developed so that the heating rates can be minimized to satisfy material characteristics. This in turn, effects the structural design of the aerobrake. Materials are also considered which would have an impact on reducing the weight of the aerobrake. Of significant importance is the atmospheric characteristics of Mars which has large variations of density between day and night and also is prone to effects by storms. The integration of these effects is important in the overall goal to meet the mission.

Aerobrake Analyses

Aerobrake analyses address

- Geometric configuration (L/D, $M/C_D A$) for capture and landing
- Maneuver capabilities to satisfy range and landing requirements
- Design configurations that reduce heating rates and heat loads for prescribed trajectories
- Methods to assess impacts of the Mars atmosphere due to steady and non-steady density effects
- Structural techniques using advanced materials aimed at reducing weight
- Integration of technology to meet mission goals

STCAEM/iev/31May90

Why Aerobraking?

Definition: Using planetary atmospheres for propulsive purposes

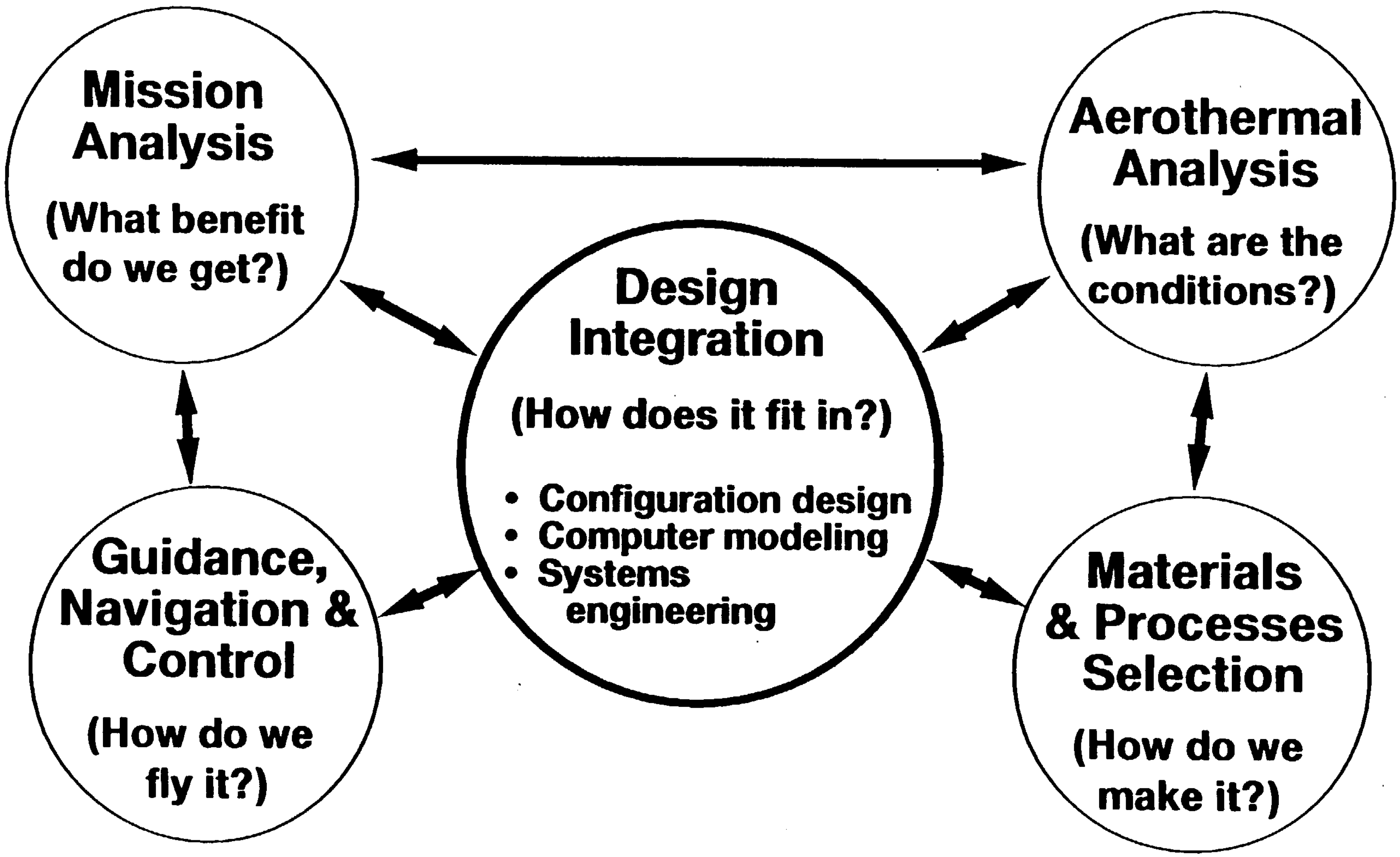
- *drag* slows vehicle, reduces propellant needs
- *lift* limits heating and g-loading, allows steering

Importance:

- Enhancing or enabling for mission modes of interest
- Major driver of vehicle design

Type of Propulsion / Mission Phase	Mars Capture	Mars Landing	Earth Capture (Lunar)	Earth Capture (Mars)	Earth Entry
Chemical all-propulsive		X			X
Chemical aerobraked	X	X	X	X	X
Electric propulsion		X	X		X
Nuclear rocket		X	X	X	X
Cycler orbits	X	X	----	X	X

Aerobraking: An Interdisciplinary Effort



Coordination among many specialties is required

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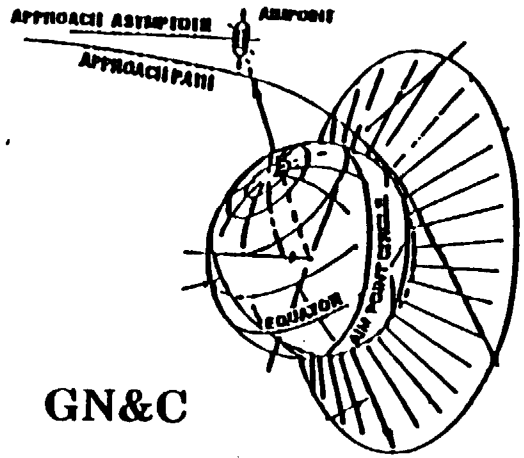
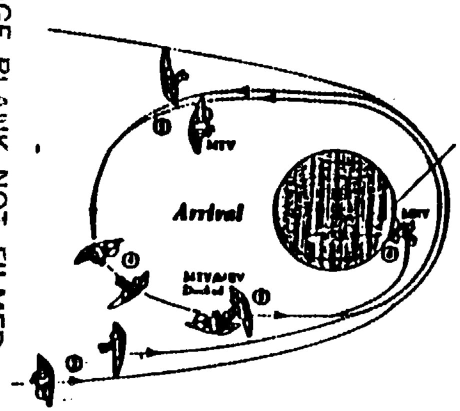
Aerobrake Analyses

Trajectory and Performance

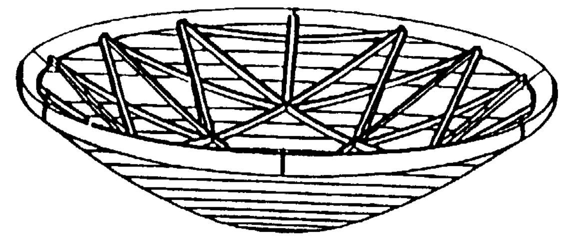


GN&C

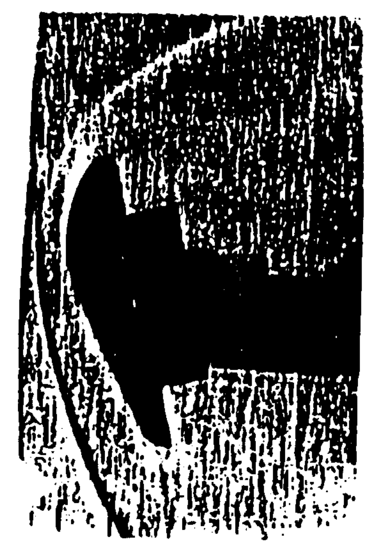


Aerocapture Descent and Landing

Structures



Aerothermal



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ADVANCED CIVIL SPACE SYSTEMS

0.5 L/D AEROBRAKE

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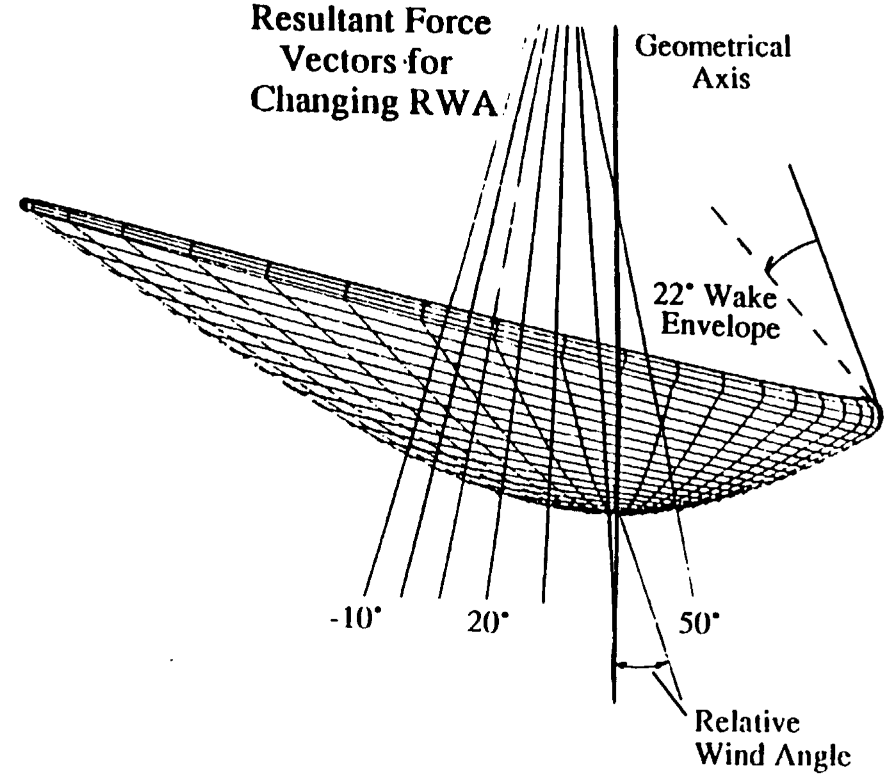
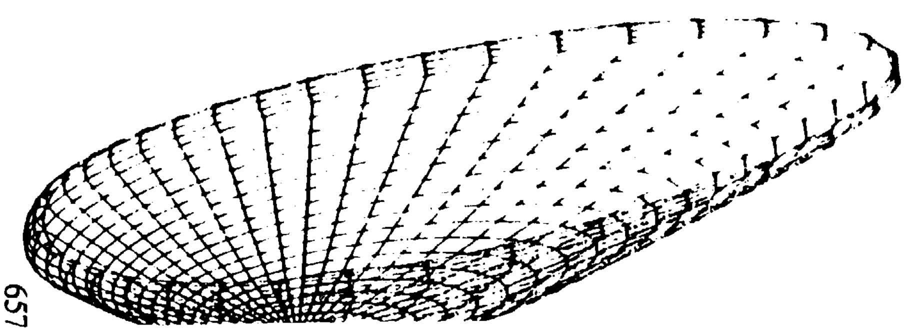
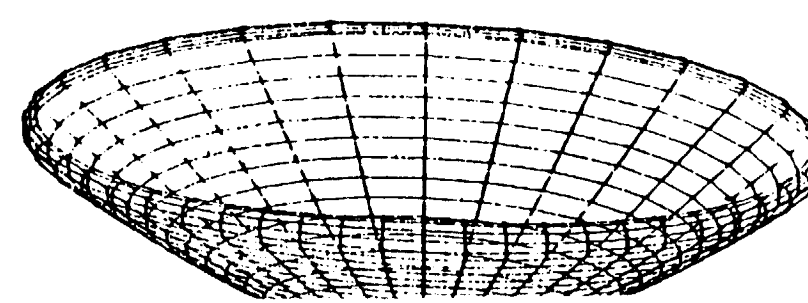


Values for Shape Parameters

Semimajor Axis Ratio	2.00
Eccen. of Body of Revol.	1.60
Eccen. of Cutting Cyl.	0.40
Truncation/SMA ratio	0.00
Lip Radius/SMA ratio	0.05
Lip Taper Ratio	0.30
Normalized x length	4.07
Normalized Plan Area	11.8
Normalized Surface Area	15.4

Angle	CL	Cd	L/D	Moment Arm
-10	0.1558	1.5251	0.1021	0.9105
0	0.3307	1.3956	0.2370	0.7861
10	0.4434	1.1902	0.3725	0.6489
20	0.4790	0.9434	0.5077	0.4913
30	0.4439	0.6983	0.6356	0.3042
40	0.3614	0.4867	0.7425	0.0795
50	0.2622	0.3215	0.8155	-0.1729





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High L/D Aerobrake - Aerodynamic characteristics of the high L/D aeorbrake, based on Modified Newtonian impact theory

ADVANCED CIVIL SPACE SYSTEMS

HIGH L/D AEROBRAKE

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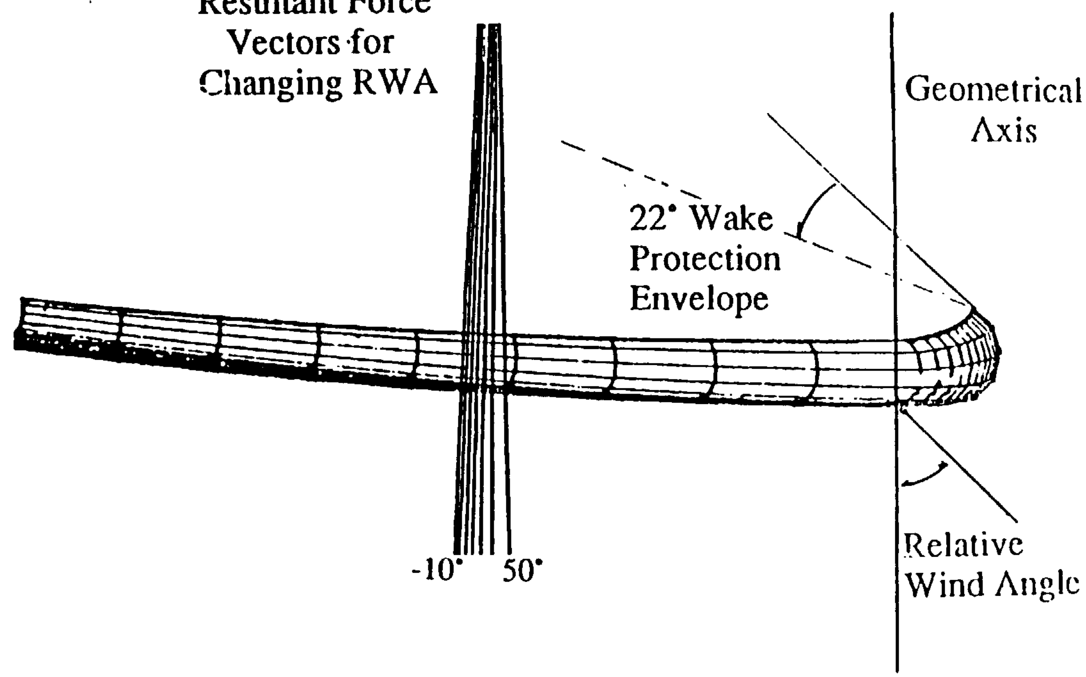
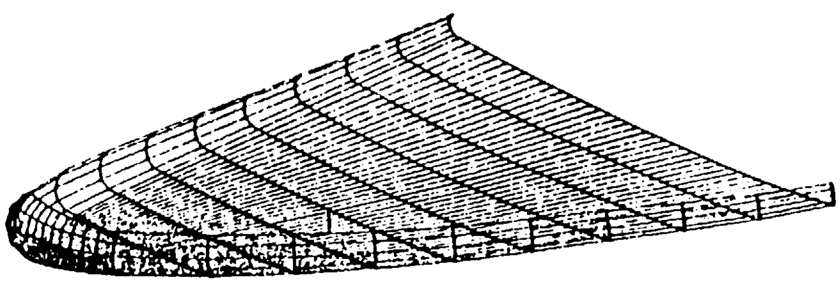
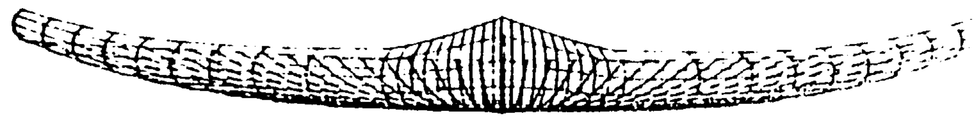


Values for Shape Parameters

Semimajor Axis Ratio	2.00
Eccen. of Body of Revol.	3.00
Eccen. of Cutting Cyl.	1.03
Truncation/SMA ratio	0.50
Lip Radius/SMA ratio	0.03
Lip Taper Ratio	0.30
Normalized x length	1.120
Normalized Plan Area	0.851
Normalized Surface Area	0.988

Angle	CL	Cd	L/D	Moment Arm
-10	-0.2113	1.7644	-0.1197	0.3894
0	0.0926	1.7960	0.0516	0.3916
10	0.3759	1.6711	0.2249	0.3954
20	0.5800	1.4132	0.4105	0.4012
30	0.6682	1.0778	0.6199	0.4112
40	0.6311	0.7267	0.8684	0.4292
50	0.4974	0.4249	1.1706	0.4650



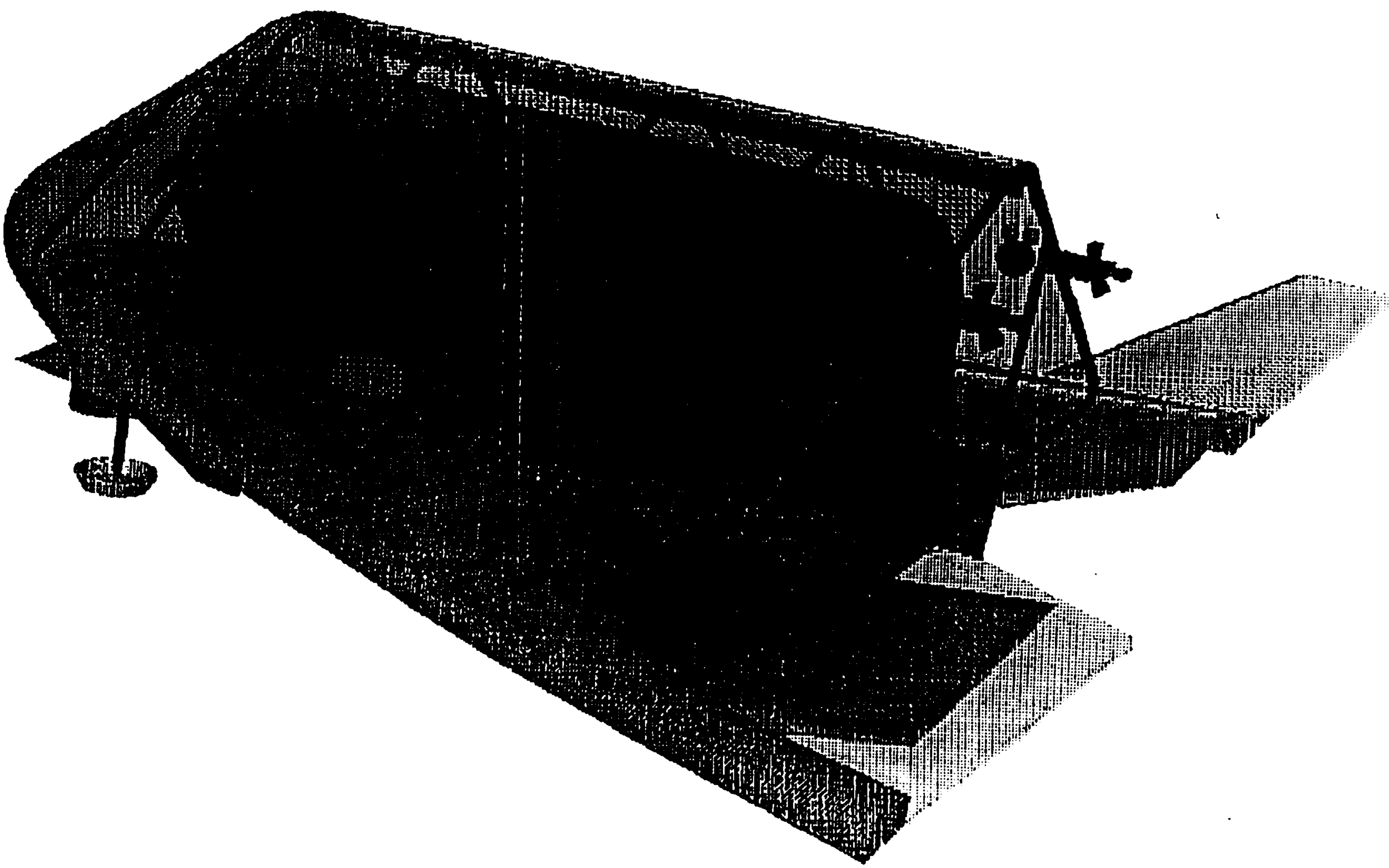


MEV CADD Models

Shown on the following pages are the reference low L/D (0.5) Mars excursion vehicle, and the high L/D (L/D 1.1) reusable MEV. The low L/D MEV acommodates a crew of 4 to 6 for a thirty day surface mission, and returns the crew to mars orbit via a separate ascent stage(MAV).

Evolution of the high L/D MEV would follow for later, more aggressive missions. The high L/D vehicle has been configured in an expendable cargo version, and in a crew/reusable version that refuels with mars surface produced propellant

High L/D MEV Configuration



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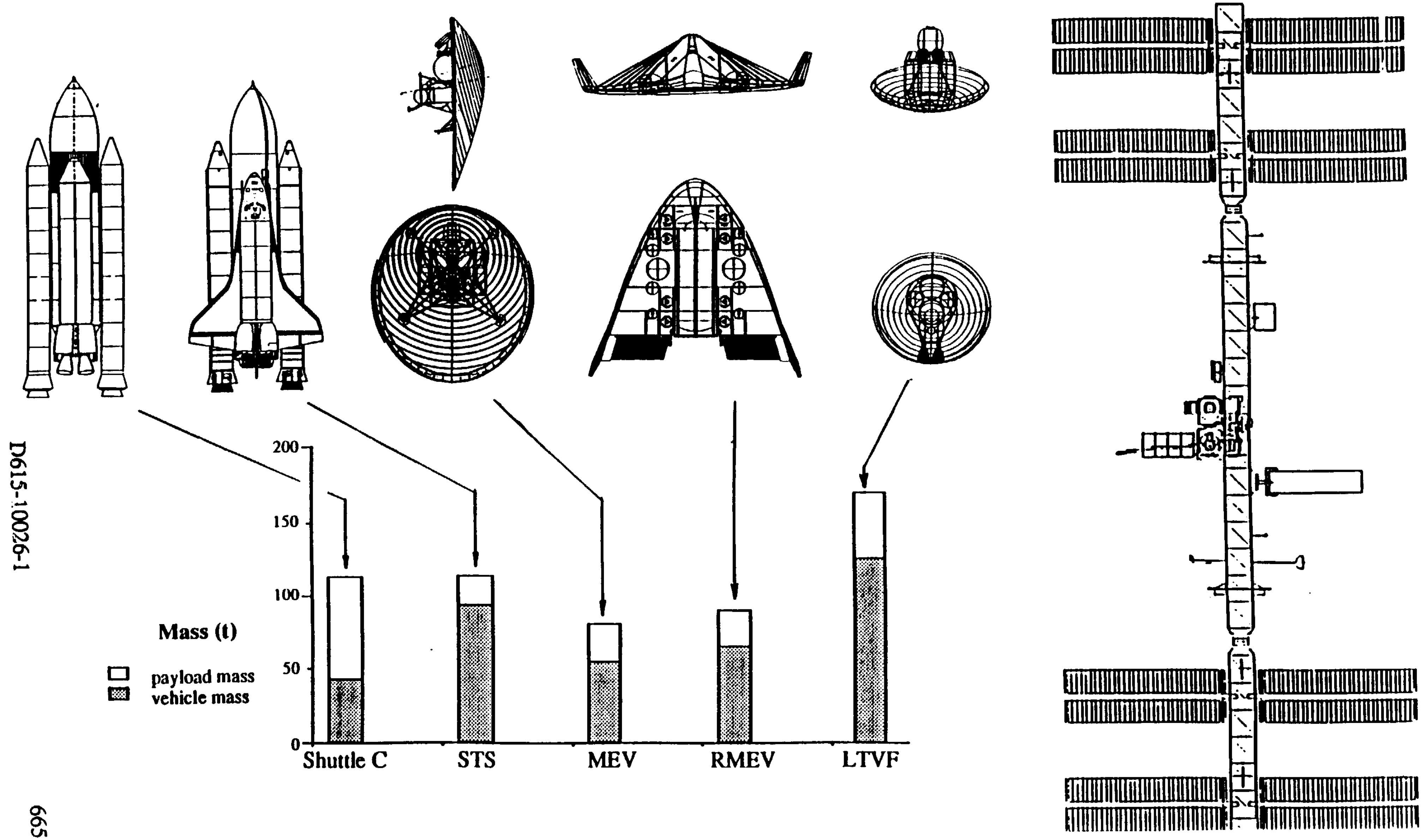
The stagnation point heating rates were calculated using the MARSIN code with a two dimensional trajectory and MarsGRAM atmosphere (high density). Te convective heat transfer was calculated for a fully catalytic wall, the radiative heat transfer was calculated using the Tauber_Sutton method and equilibrium flow. The stagnation point radius of curvature was . The range of L/D varied form 1.0 to -1.0. For each condition a fixed L/D was used. The calculations illustrate that as L/D becomes negative the heating rate decreases.

STCAEM Aerobraked Vehicles Comparison

Shown on the facing page are the representative aerobraked transfer vehicle overall size and mass, as compared with shuttle launch vehicles, and Space Station Freedom. The SSF comparison is made to illustrate the difficulty of assembling these vehicles at Space Station.

STCAEM Aerobraked Vehicles Comparison



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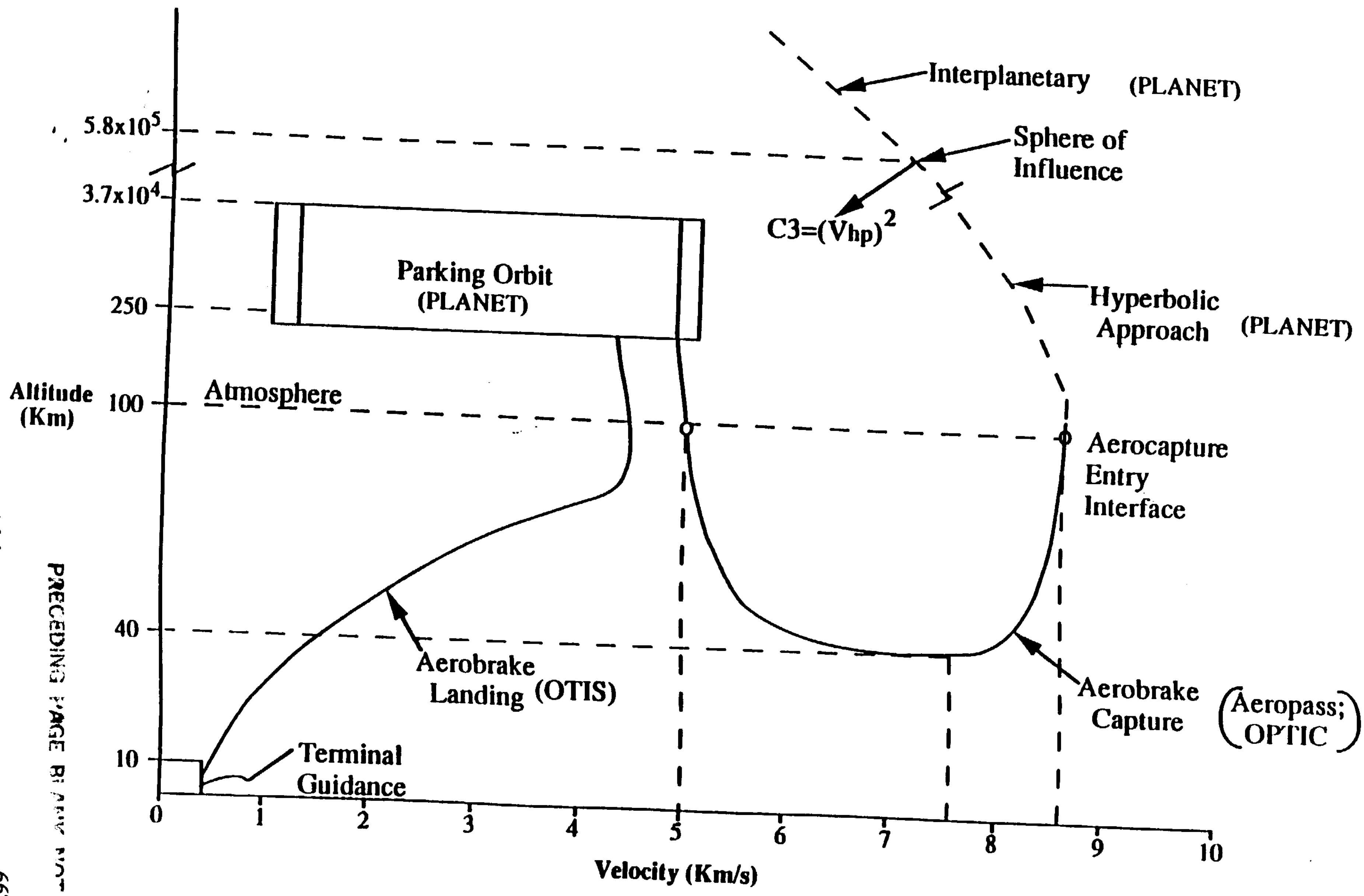
Aerobrake Capture and Landing



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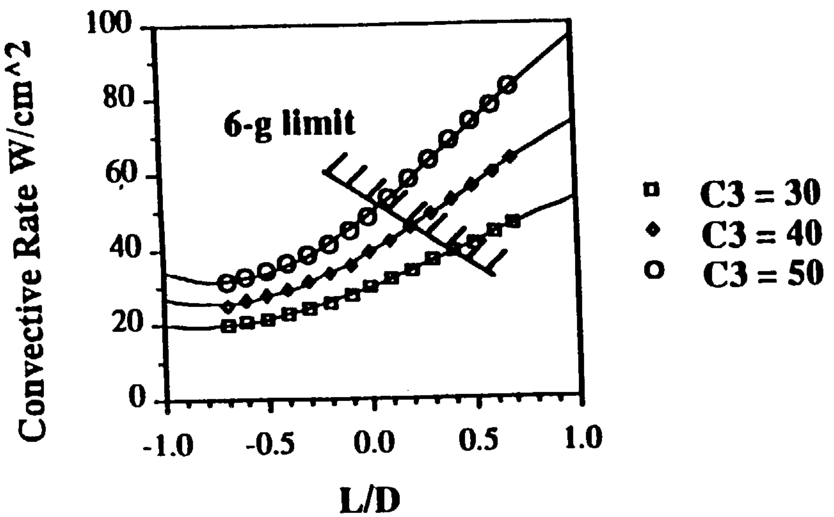
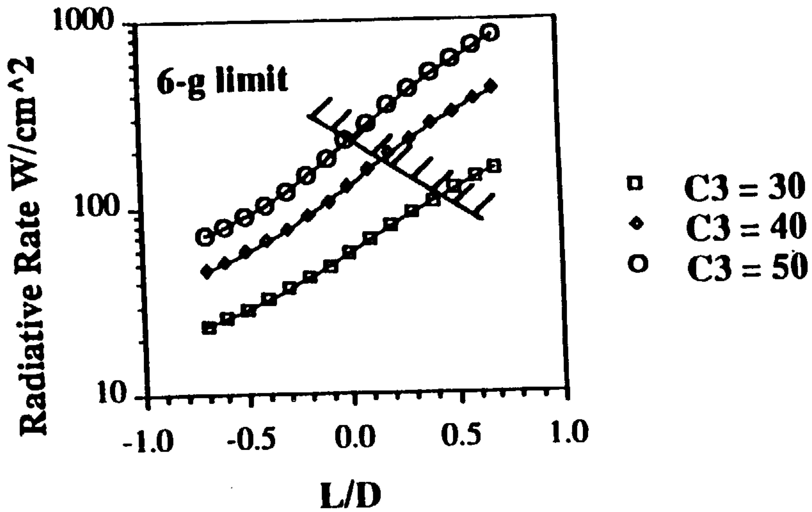
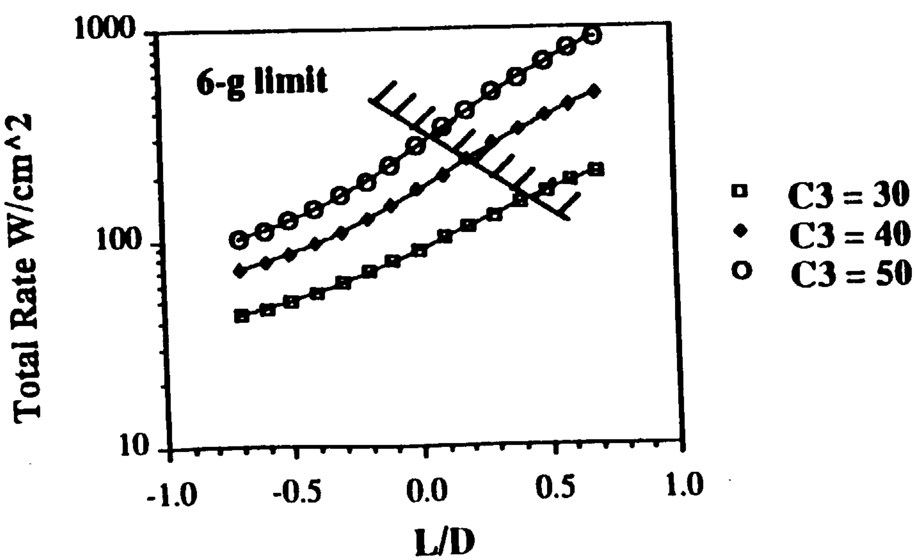
The stagnation point heating rates were calculated using the MARSIN code with a two dimensional trajectory and MarsGRAM atmosphere (high density). Te convective heat transfer was calculated for a fully catalytic wall, the radiative heat transfer was calculated using the Tauber_Sutton method and equilibrium flow. The stagnation point radius of curvature was . The range of L/D varied form 1.0 to -1.0. For each condition a fixed L/D was used. The calculations illustrate that as L/D becomes negative the heating rate decreases.

Stagnation Point Heating Rates
Mars Aerocapture - MTV







Stagnation Point
Radius of 13 m

Ballistic Coefficient
394 Kg/m^2

Radiation:
Tauber-Sutton Method

MarsGRAM
Atmosphere Hi
Density

Mars Aerocapture Stagnation Point Heating - MTV

Aerocapture trajectory was computed with the MARSIN code.
Initial conditions are given as follows:
- MTV hyperboloid aerobrake
- L/D = 0.5
- Flight averaged L/D = 0
- Approach C3 = 30 sq km/sq sec
- Entry altitude = 150 km
- Entry velocity = 7.4 km/sec
- Ballistic coefficient = 394 kg/sq m
- MarsGRAM high density (winter solstice) for 2016

Mars Aerocapture Stagnation Point Heating - MTV

ADVANCED CIVIL SPACE SYSTEMS

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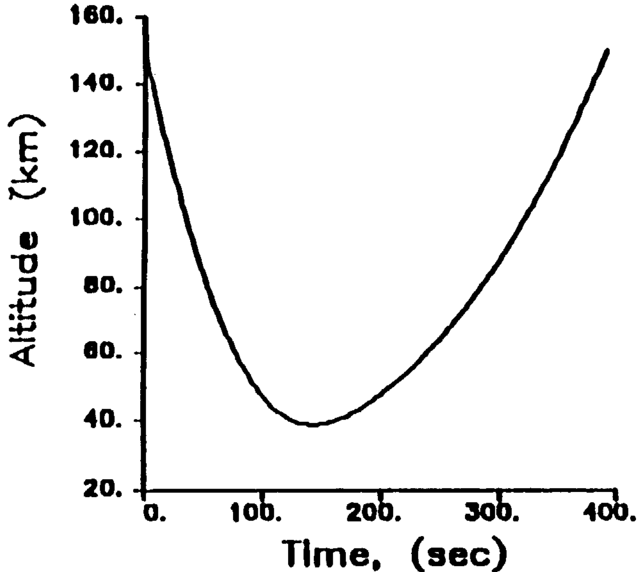

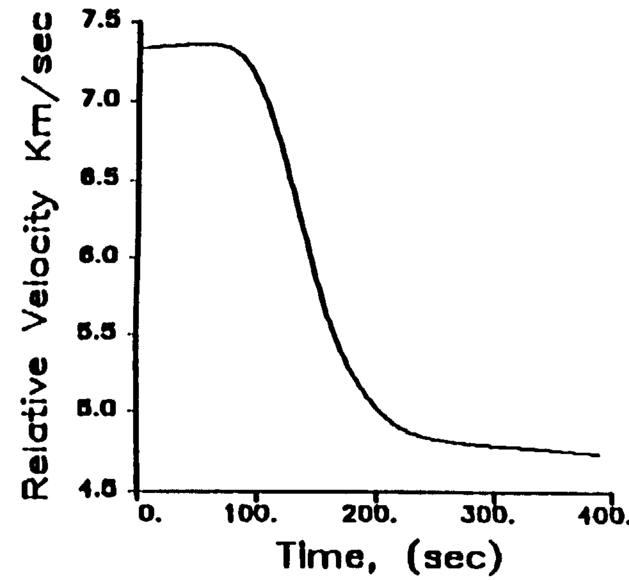

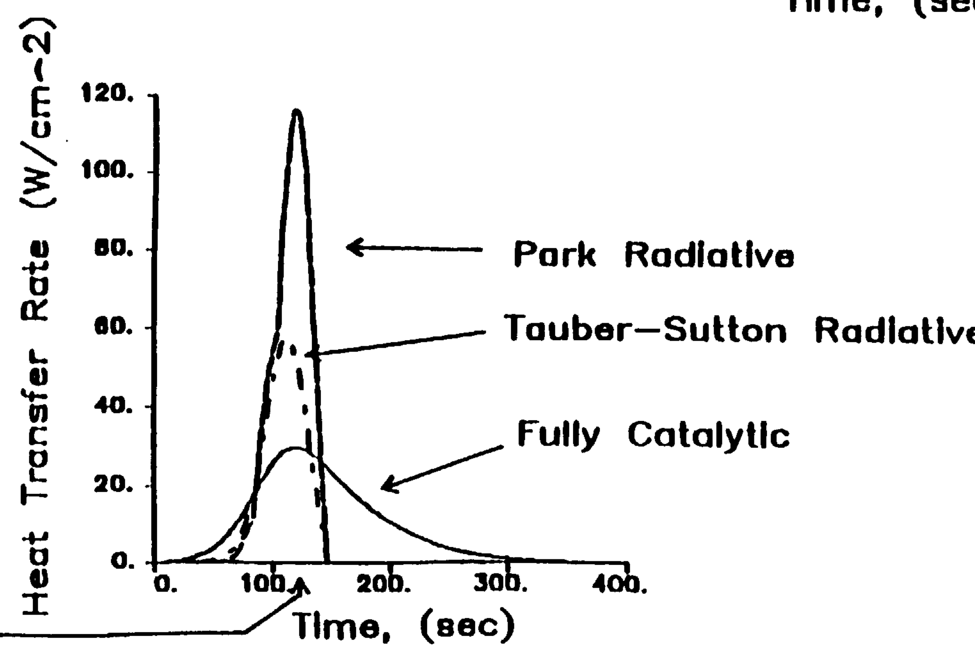

Maximum Total Heat Flux at 120 sec

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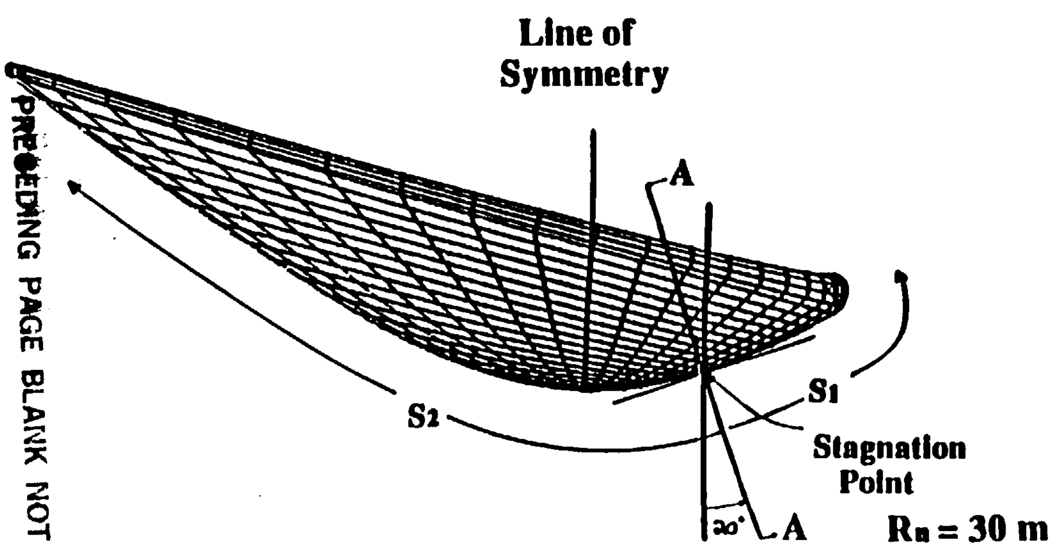
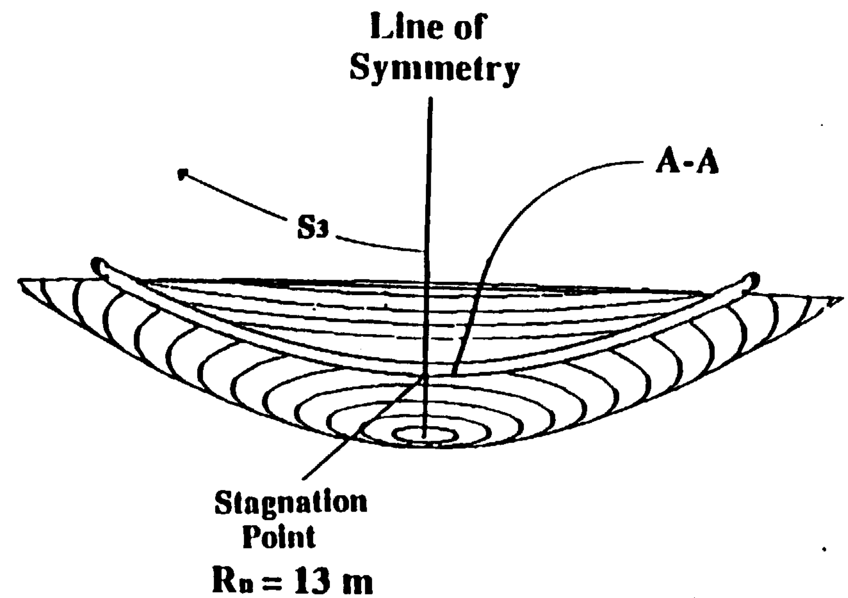
Streamlines for Hyperboloid Aerobrake

Assumed Streamlines S_1, S_2, S_3

For an angle of attack of 20°

L/D = 0.5





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The following three charts show the pressure distribution along the fore , aft, and side streamlines based on a modified Newtonian theory, and unswept cylinder theory for the cylindrical lip.

ADVANCED CIVIL SPACE SYSTEMS

Forward Centerline Pressure Distribution

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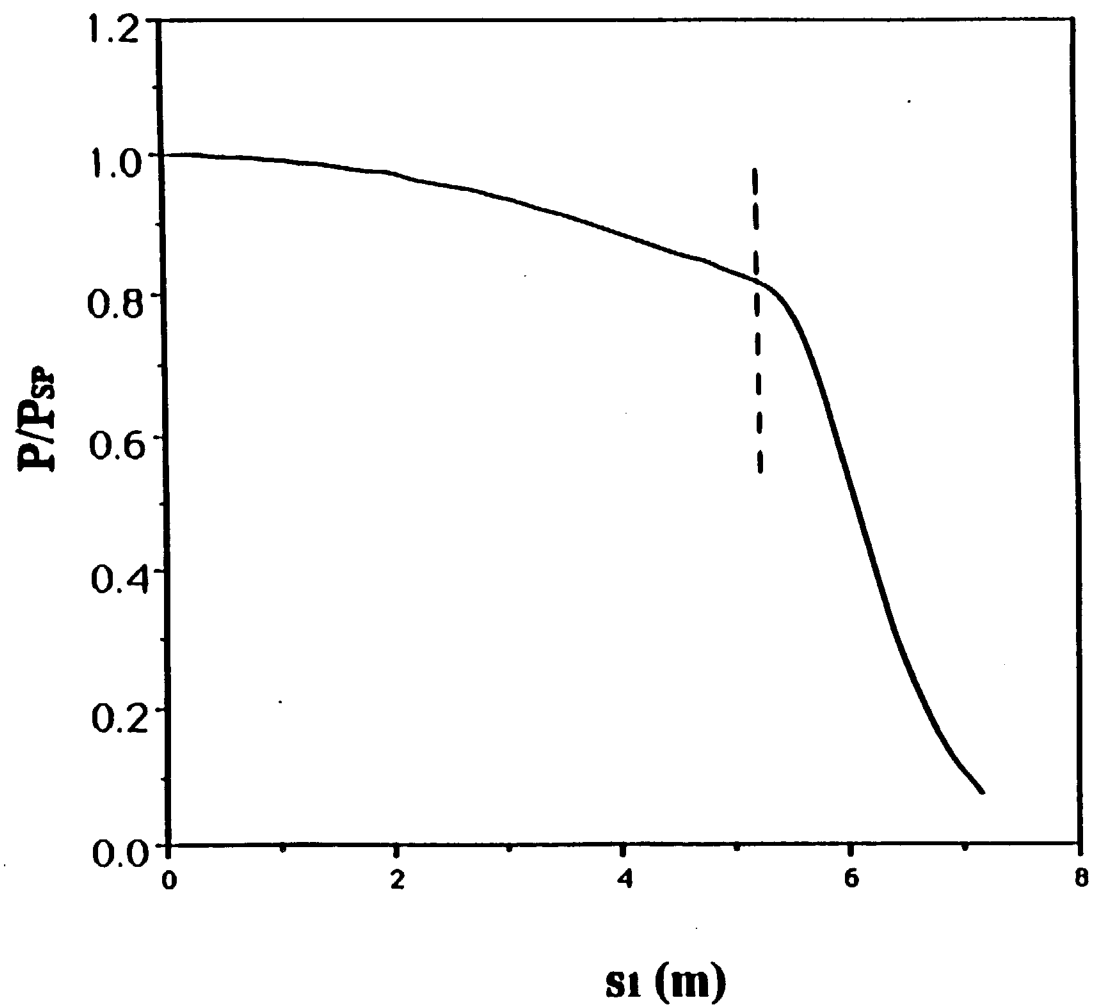
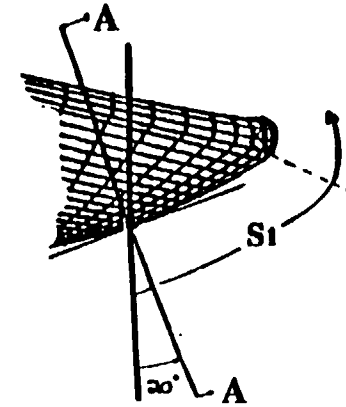
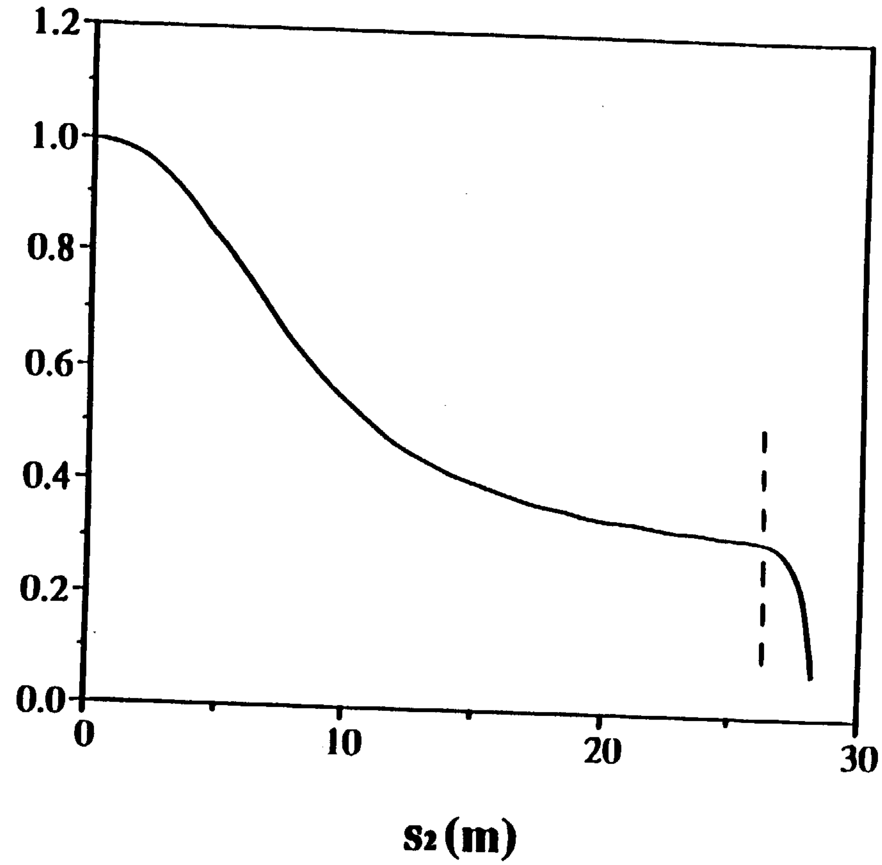
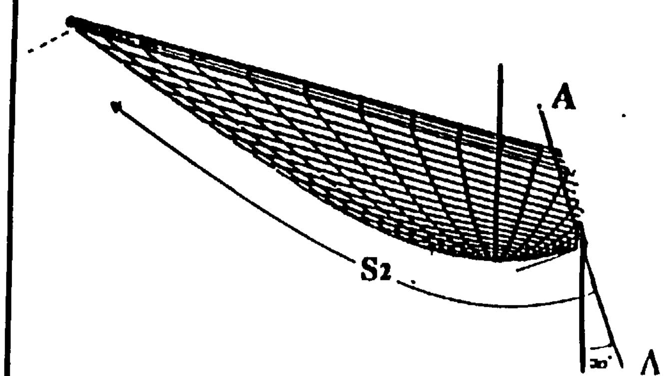




P_{SP} = Stagnation Point Pressure

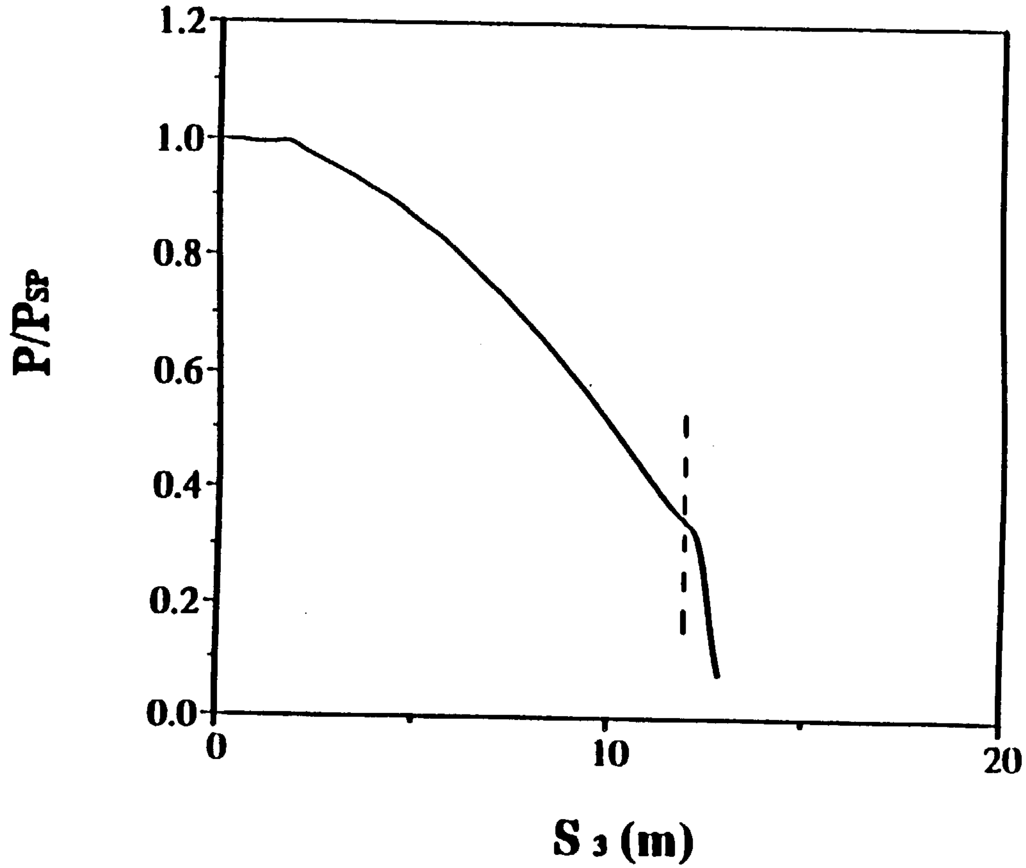
Aft Centerline Pressure Distribution

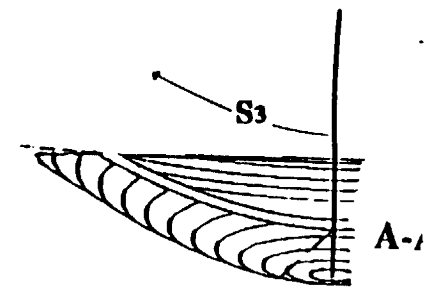




P_{SP} = Stagnation Point Pressure

Side Streamline Pressure Distribution





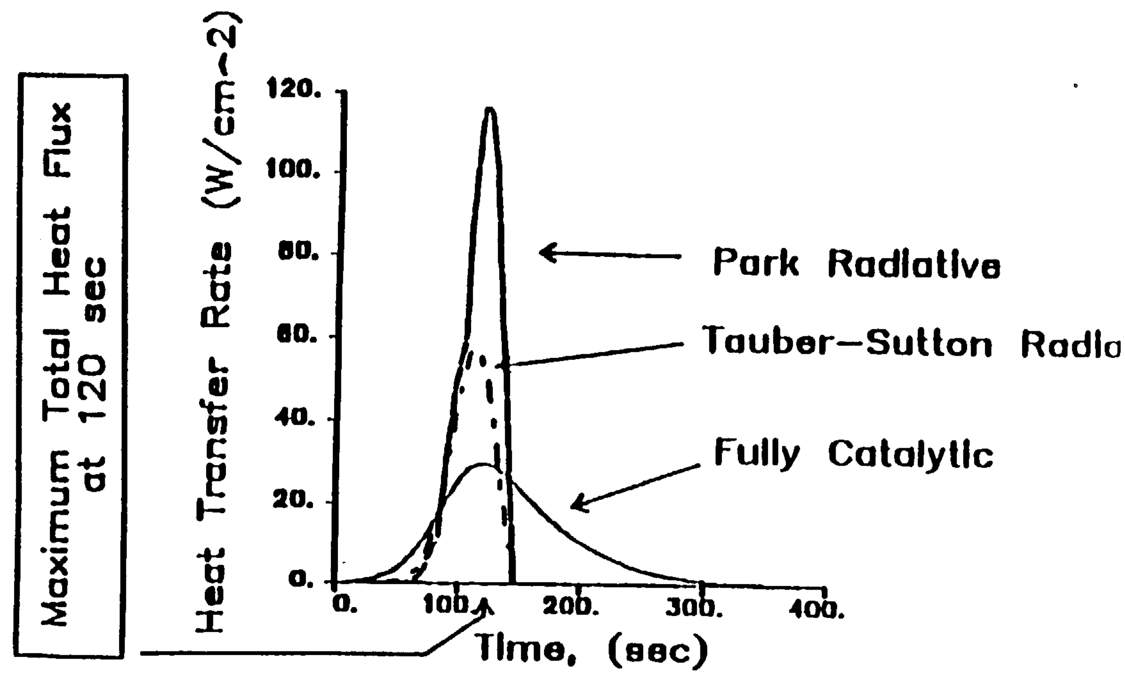
P_{SP} = Stagnation Point Pressure

GRM/2H895/AEROBRAKE STUDY/DISK 2/N/141-0/7:00A

The following chart delineates the aerothermal conditions at the stagnation point for maximum heating along the aerocapture trajectory.

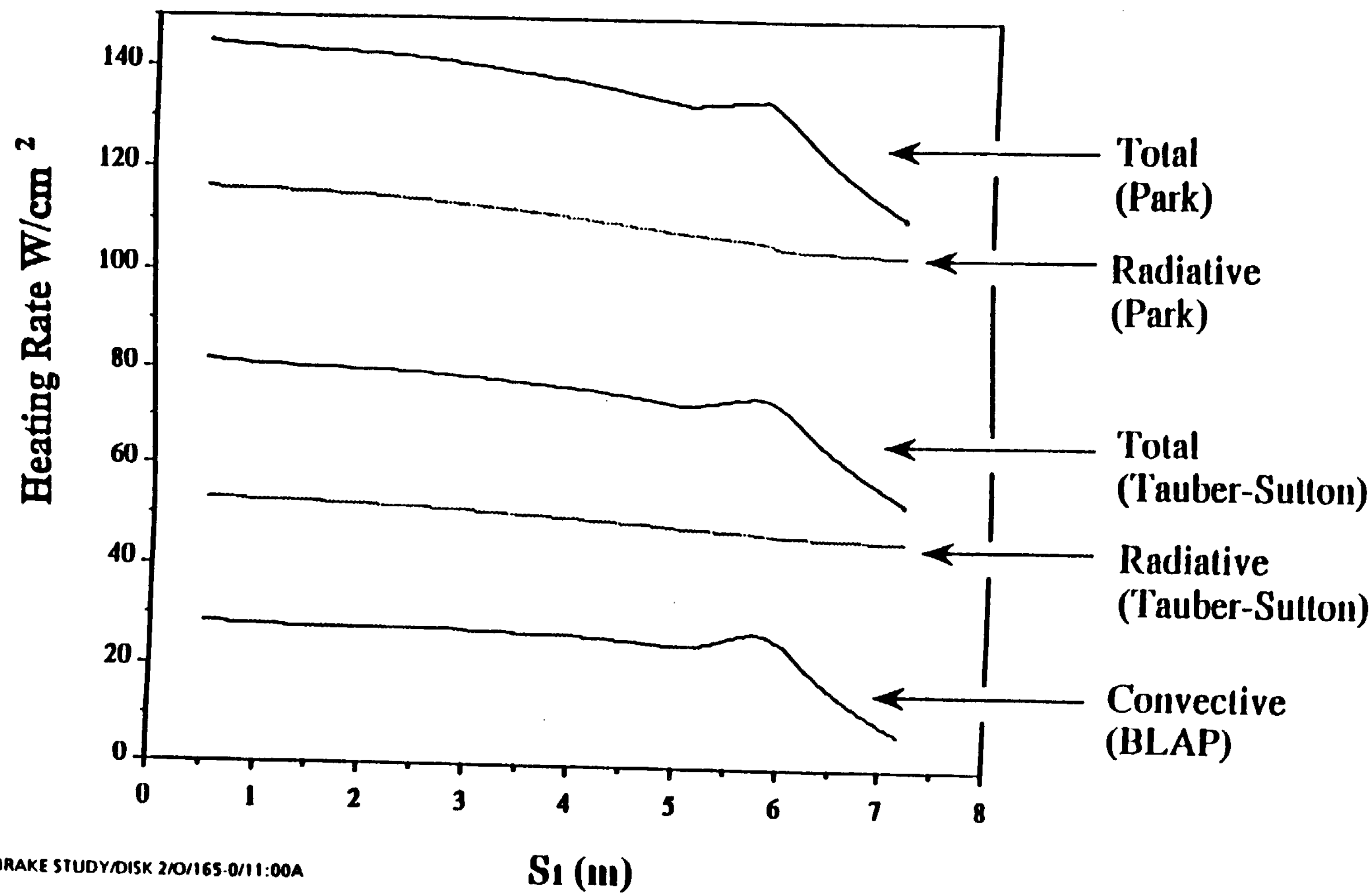
Conditions at Maximum Stagnation Point Heating

- Time - 120 sec
- Velocity - 6.714 km/sec
- Altitude - 41.2 km
- Density - $4.78 \times 10^{-7} g/cm^2$
- Radiative
  - Park - 116 w/cm^2
  - Tauber-Sutton - 53 w/cm^2
- Convective - 30 w/cm^2
- Equilibrium wall temperature (emissivity = .8)
  - Park - 2300°k
  - Tauber-Sutton - 2070°k
- Maximum G-level = 3.0



For the hyperboloid aerobrake of approximately 30m length, the heating rate was calculated using the boundary layer analysis program (BLAP) for the convective heat transfer with the radiative heat transfer being calculated using the Park method and the Tauber-Sutton method. For the forward stream line heating, the radiative heat transfer using the Park method is approximately twice that of the Tauber-Sutton method. No turbulent transition was assumed for the calculation. The heating rate at the stagnation point is approximately 146 w/cm^2 using the Park method and approximately 80 w/cm^2 for the Tauber-Sutton method.

Forward Streamline Heating



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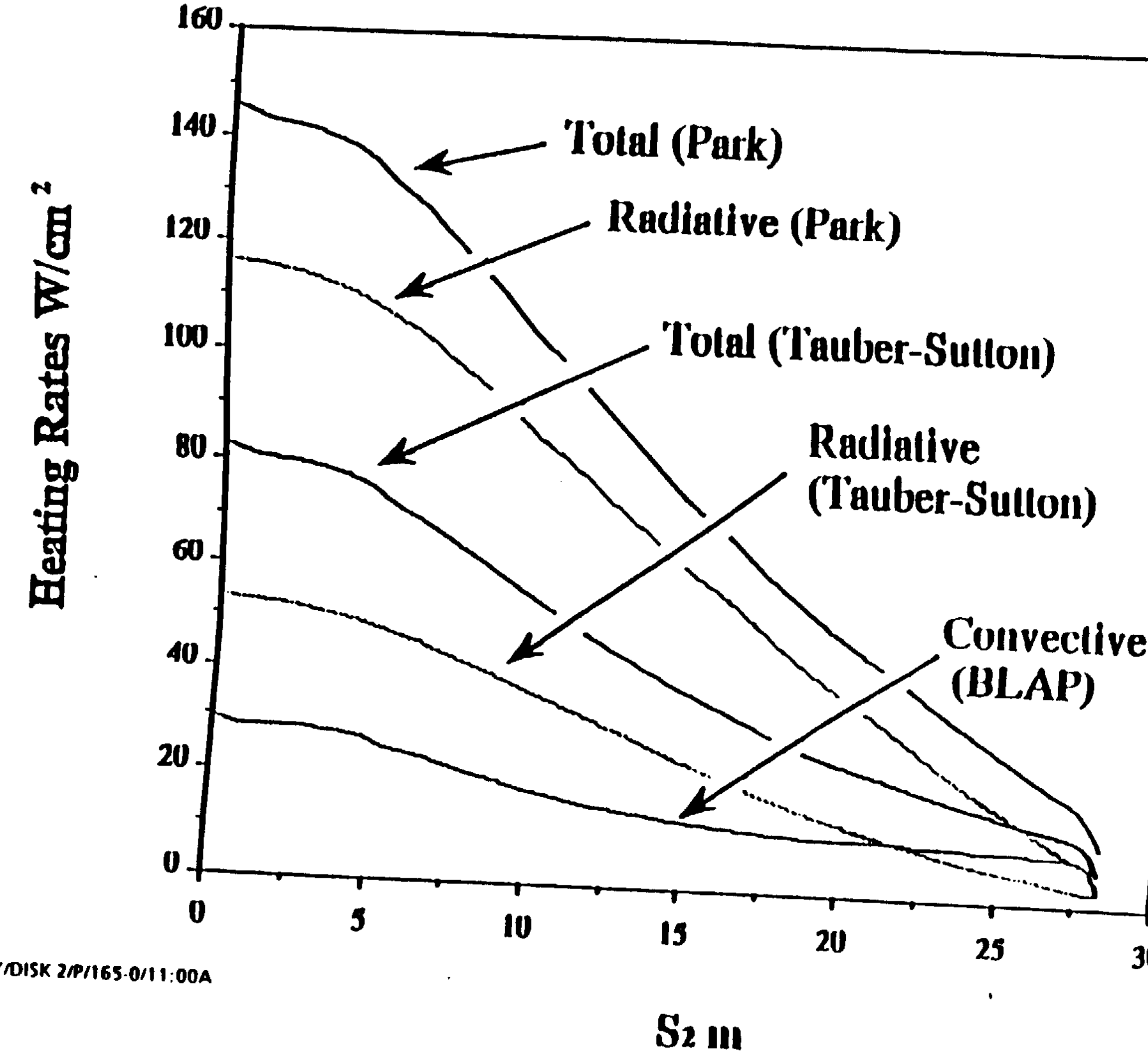
Aft Streamline Heating



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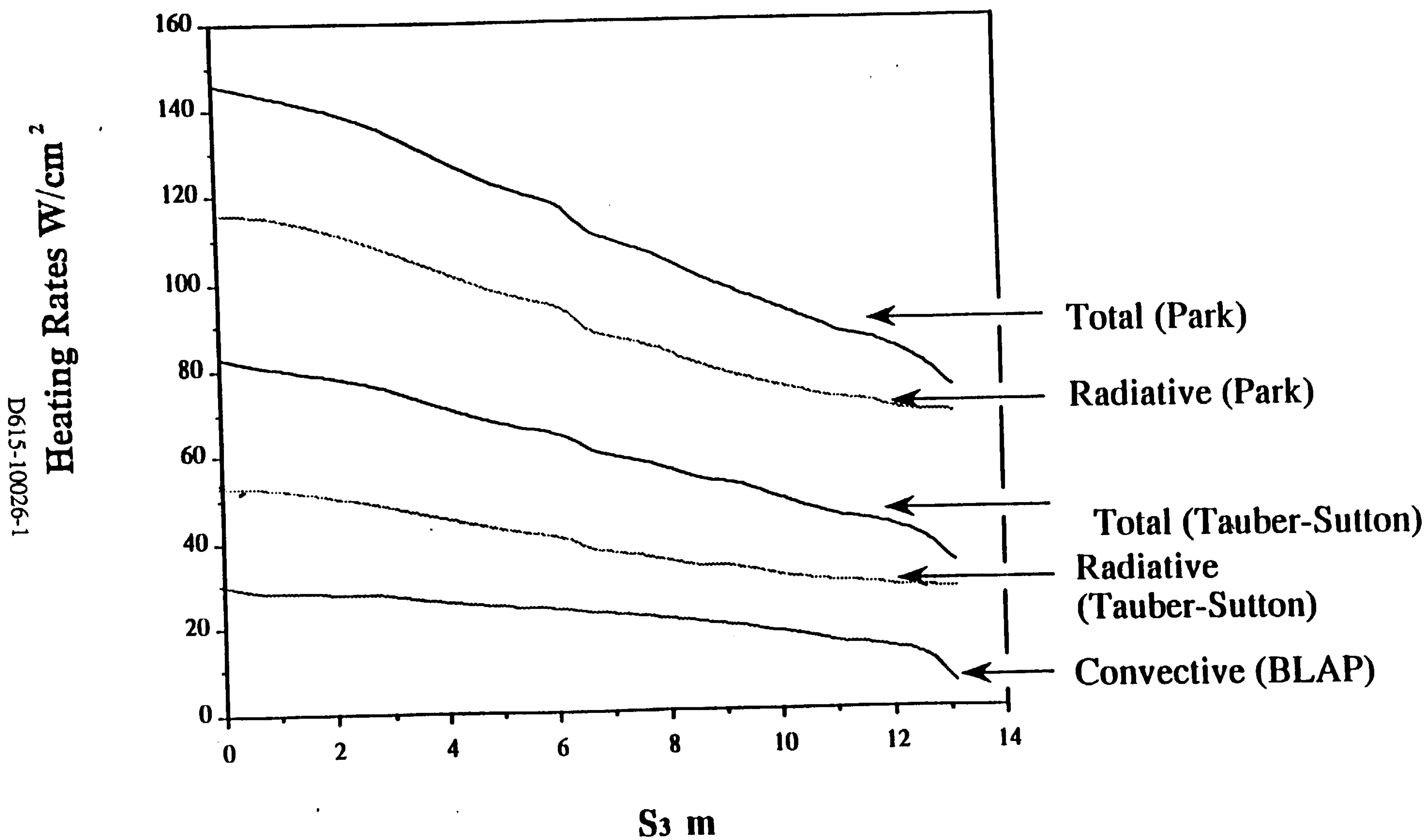
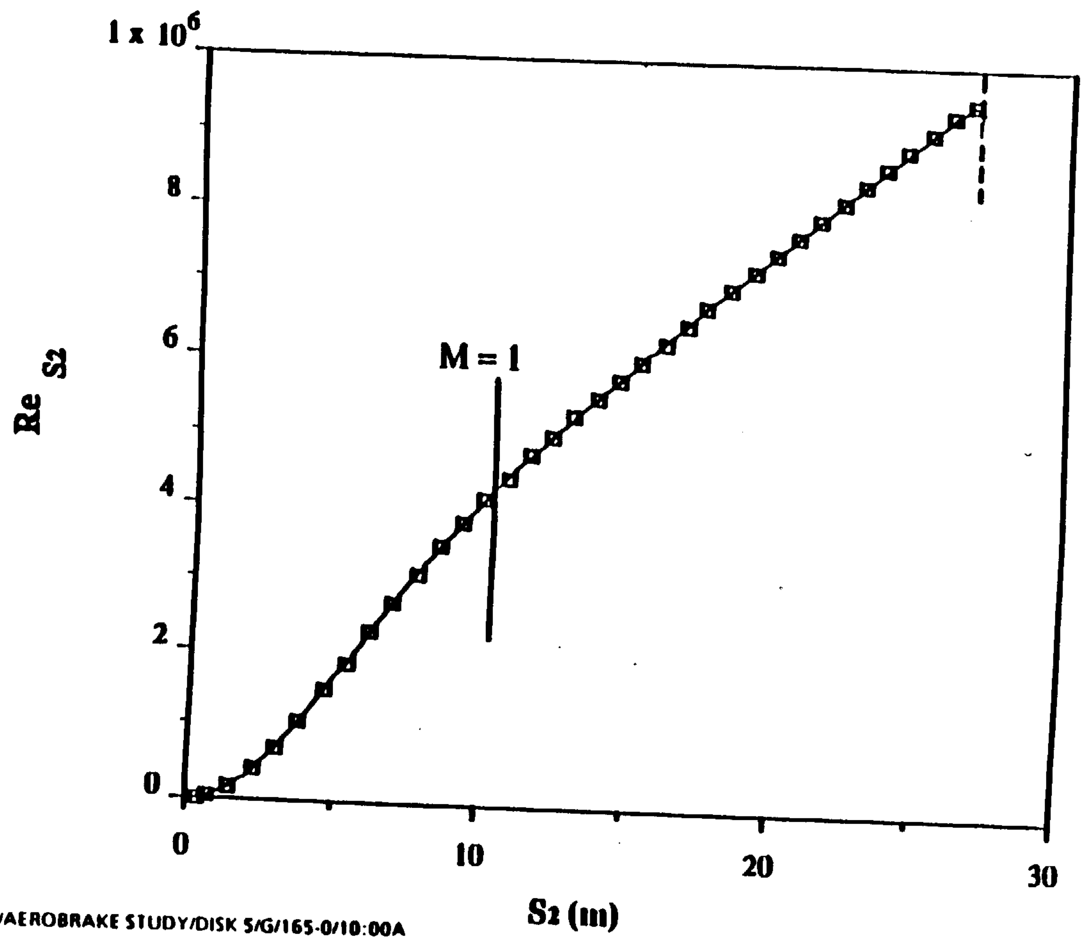
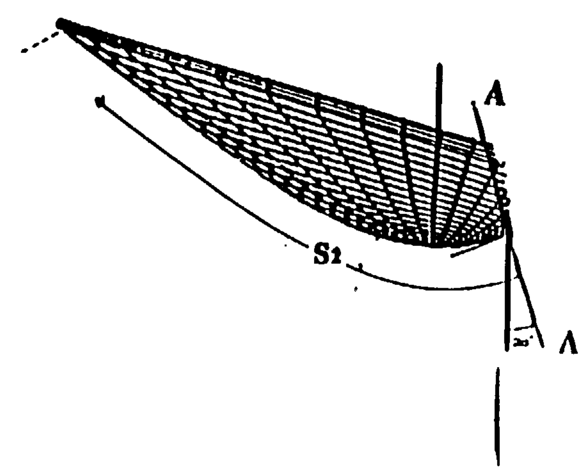
Side Streamline Heating



The Reynolds number was calculated for the aft stream line of approximately 30 meters. The Reynolds number was calculated based upon conditions behind the shock at the edge of the boundary layer utilizing the BLAP program. The calculations are for the highest heating rate at 120 sec. with a velocity of 6.7 km/sec at an altitude of 41.2 km. and a C3 of 30 km^2/sec^2. The local Reynolds number at the rear of body is less than one million.

...ds Number Along Aft Streamline





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Mars Aerocapture Trajectory
$C3 = 30\ km^2/sec^2$

This composite chart shows the atmospheric penetration, the reduction in relative velocity and the heating rate, both convective and radiative, over the 400 seconds after atmosphere encounter of the L/D 0.5 aerobrake at an entry C3 of 30. The high density winter solstice conditions were used for the worst case heating conditions for this set of entry conditions.

ADVANCED CIVIL SPACE SYSTEMS

Mars Aerocapture Trajectory C3 = 30 km^2/sec^2

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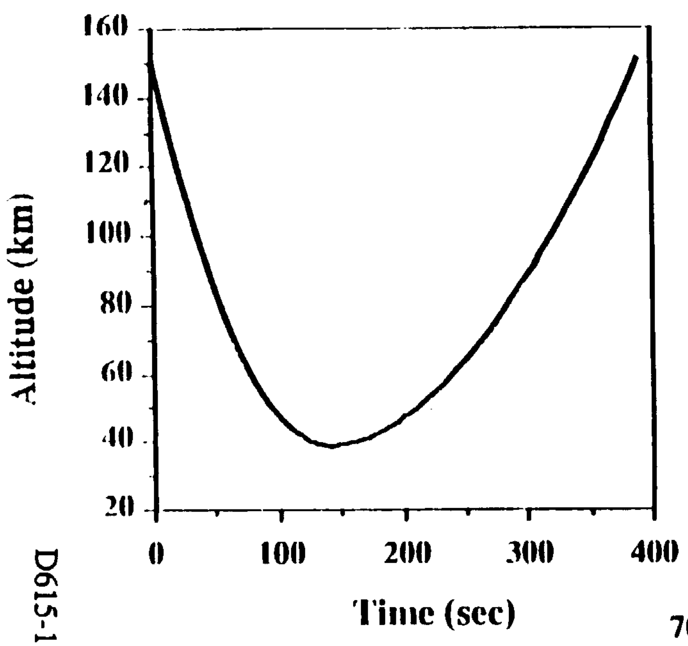
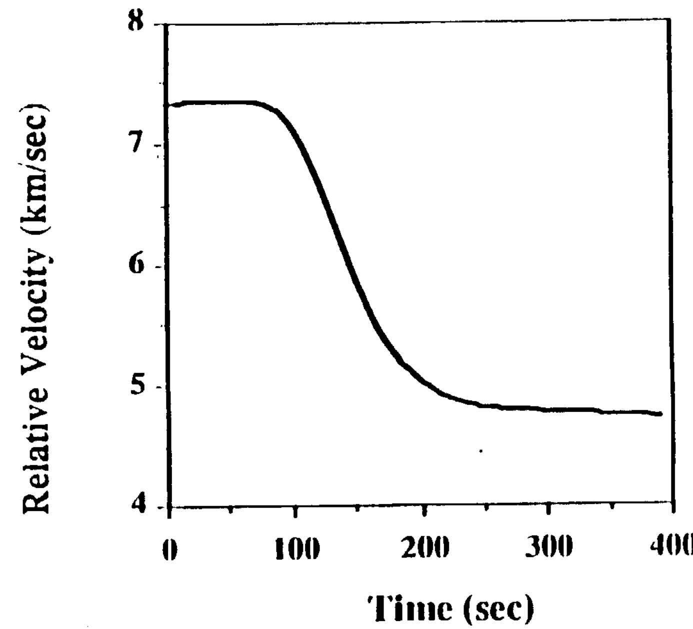
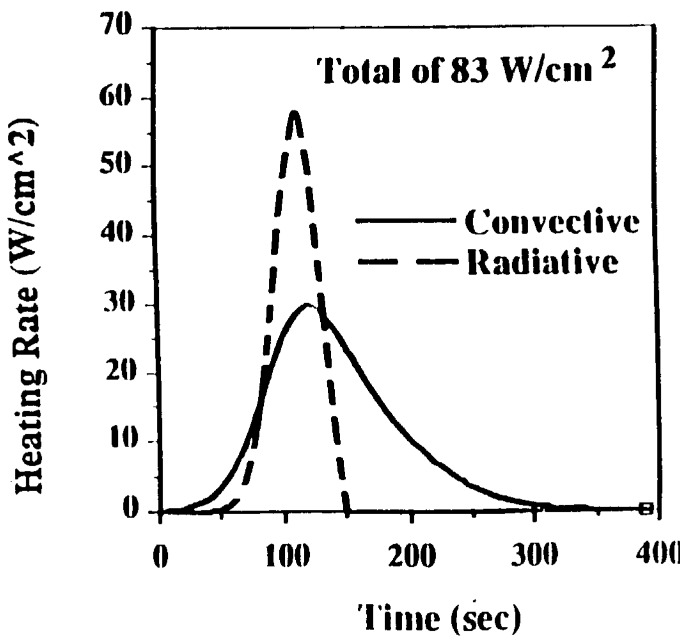






Nose Radius 13 meters

Entry Velocity 7.4 km/sec

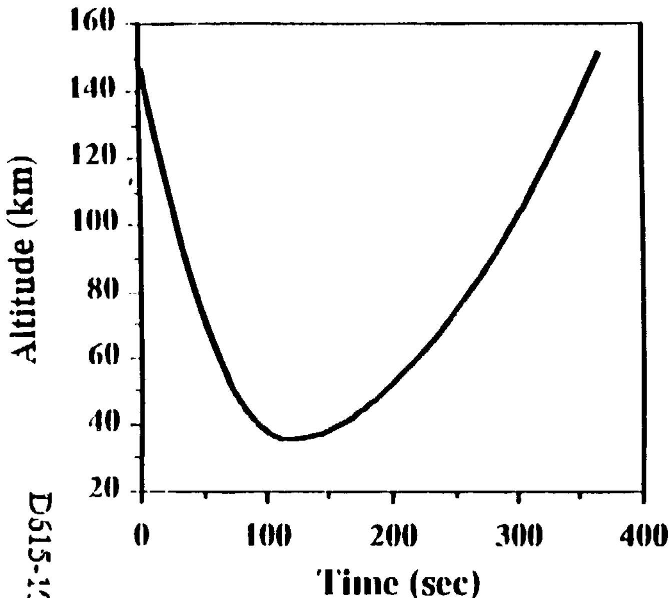
Entry Altitude 150 km

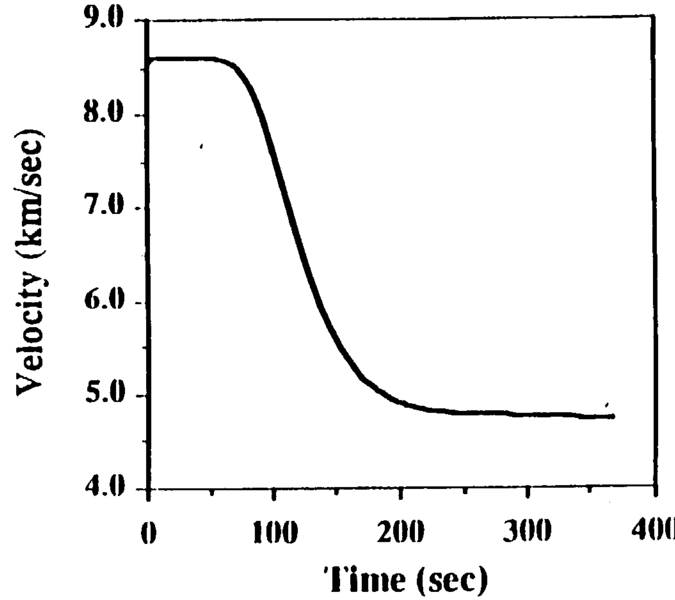
MarsGRAM High Density (2016 Winter Solstice)

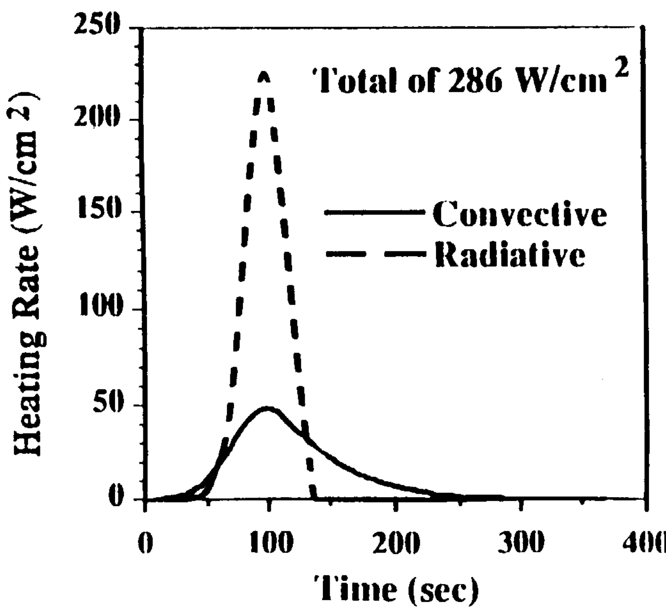
Mars Aerocapture Trajectory
C3 = 50 km^2/sec^2

This composite chart shows the atmospheric penetration, the reduction in relative velocity and the heating rate, both convective and radiative, over the 400 seconds after atmosphere encounter of the L/D 0.5 aerobrake at an entry C3 of 50. The high density winter solstice conditions were used for the worst case heating conditions for this set of entry conditions.

Mars Aerocapture Trajectory
$C3 = 50\ km^2/sec^2$







Nose Radius 13 meters

Entry Velocity 8.6 km/sec

Entry Altitude 150 km

MarsGRAM High Density
(2016 Winter Solstice)

Mars Aerocapture Comparison C3 30 & 50

This is a pictorial comparison of the temperature distribution on the L/D 0.5 aerobrake at peak heating during the aerocapture maneuver. With an increase in entry velocity of 1.2 km/sec (C3=50) the stagnation point temperature raises an additional 760 °K and trails off to a higher trailing edge temperature higher than the stagnation point of the C3=30 conditions. The conditions at a C3=30 can be handled by near-term technology developments in reradiative material. The conditions at a C3=50 is at the projected limit of advanced reradiative technology and may require an ablative surface.

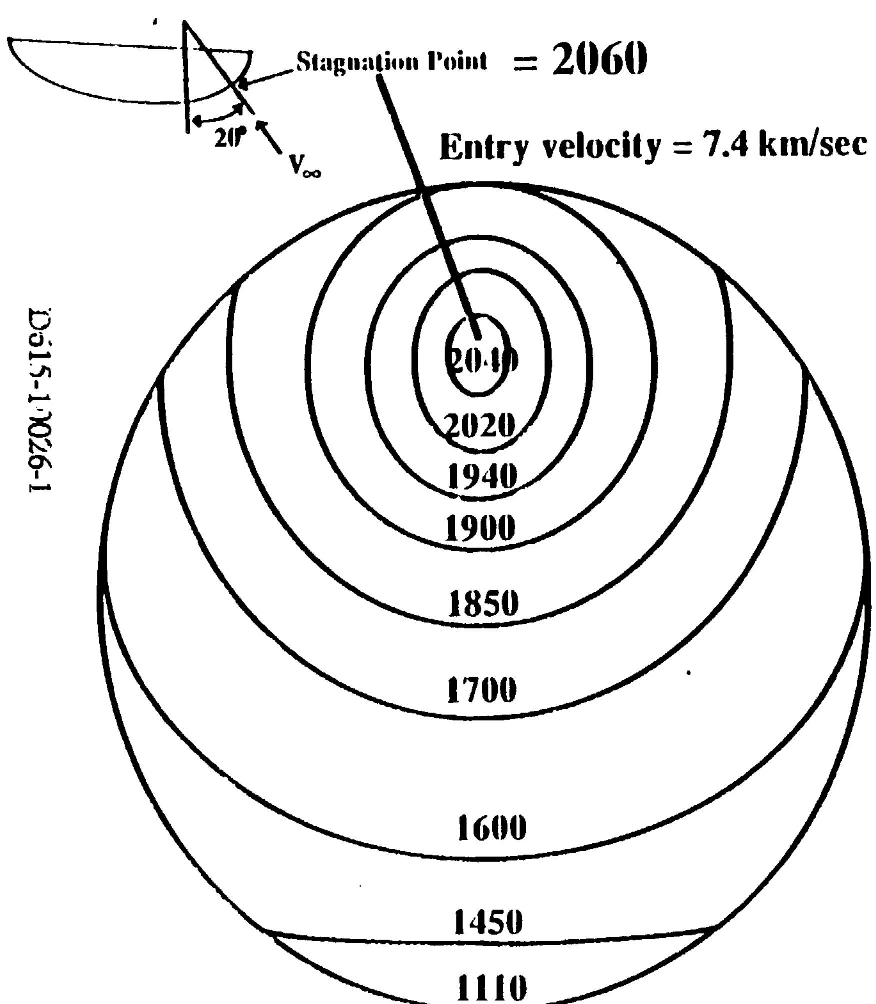
ADVANCED CIVIL SPACE SYSTEMS

Mars Aerocapture Comparison C3 30 & 50

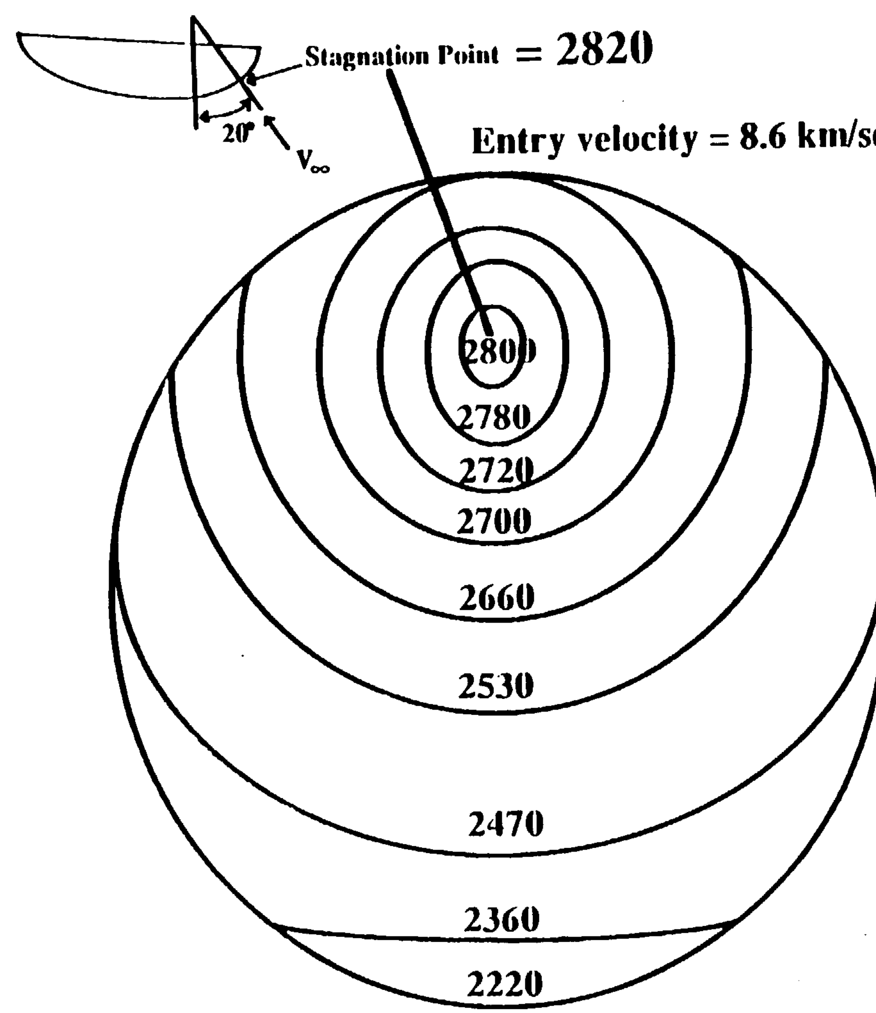
BOEING

Temperature in K

$C3 = 30\ km^2/sec^2$

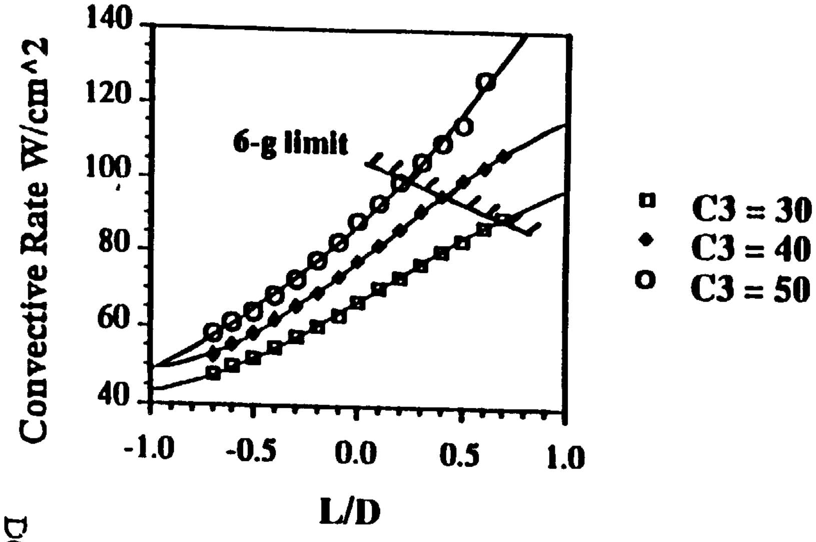


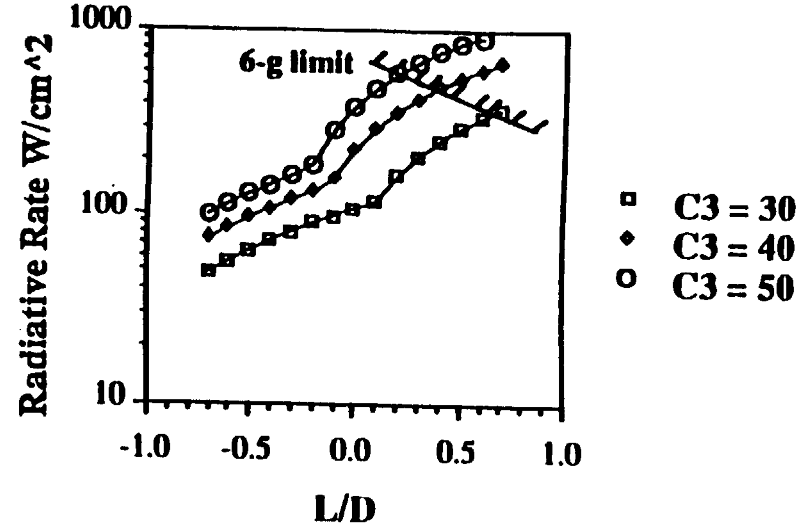
$C3 = 50\ km^2/sec^2$

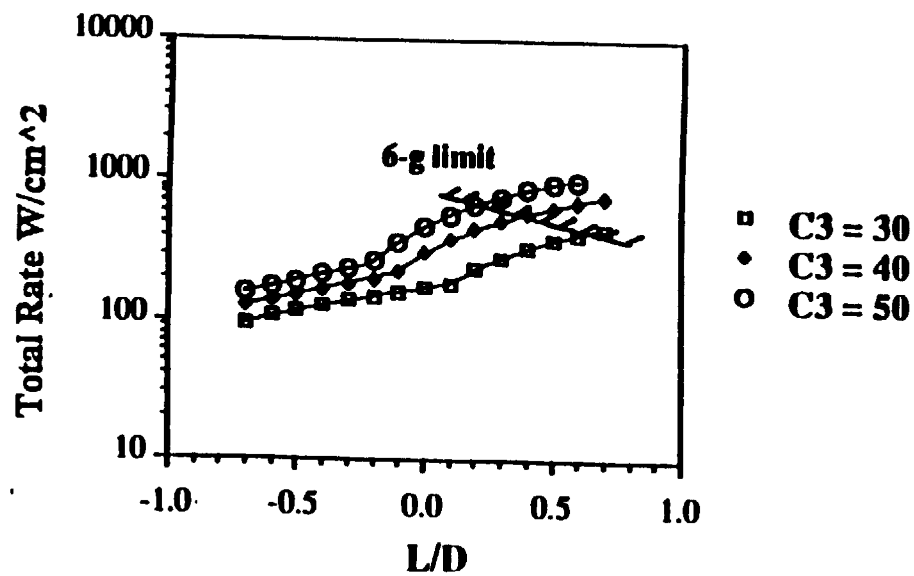


The stagnation point heating rates were calculated using the MARSRT code with a two dimensional trajectory. The convective heat transfer was calculated for a fully catalytic wall; the radiative heat transfer was calculated using the Park method and equilibrium flow. The stagnation point radius of curvature was 13 m. The range of L/D varied from 1.0. to -1.0. For each condition a fixed L/D was used. The calculations illustrate that as L/D becomes negative, the heating rate decreases.

Stagnation Point Heating Rates
Earth Aerocapture - MTV







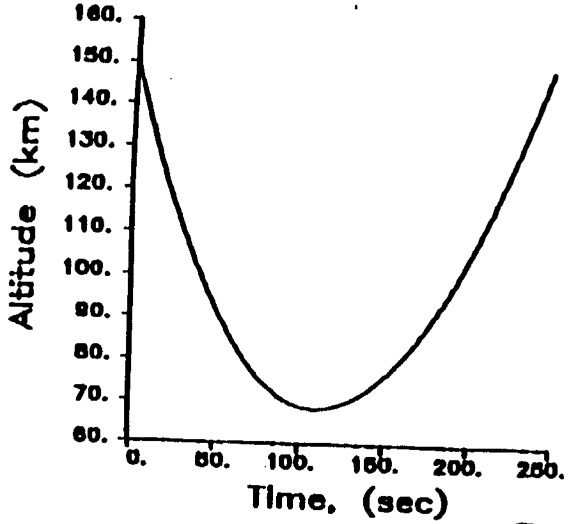
Stagnation Point
Nose Radius 13 m

Ballistic Coefficient 225 kg/m^2

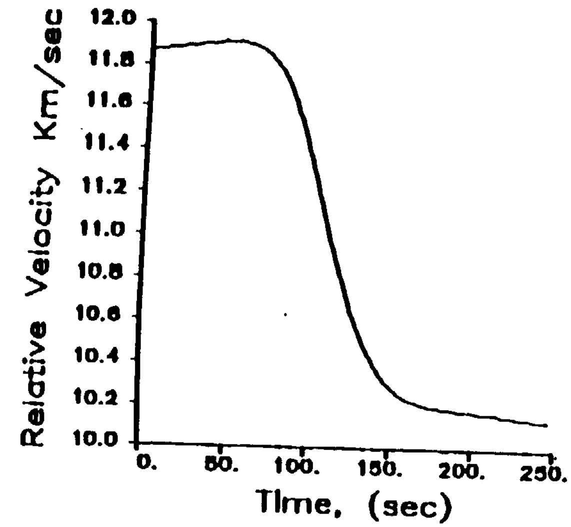
Radiative Heating
Park Method

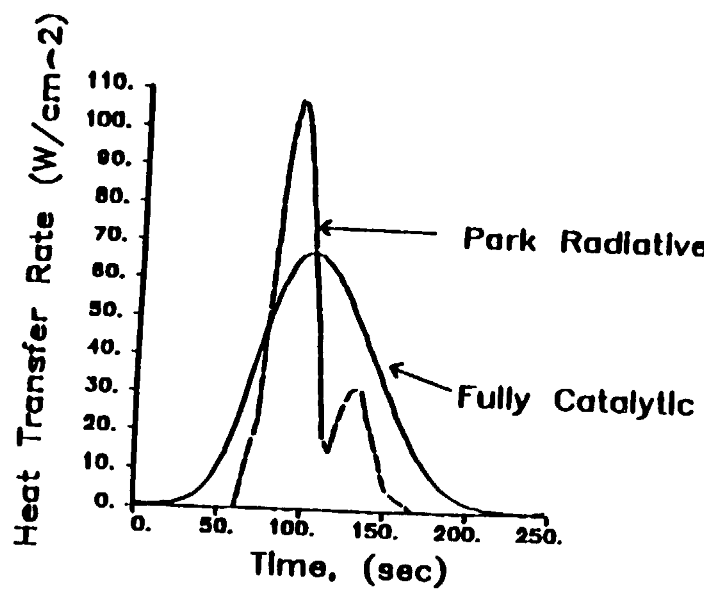
For an average L/D of 0.5 using the same relationships as previously, the Earth aerocapture total heating rate is 172 w/cm^2 using the Park method. The peak convective heating rate was about 70 w/cm^2 calculated by the BLAP code. The earth aerocapture stagnation heating rate is higher than the stagnation point heating rate for Mars which was calculated to be 146 w/cm^2.

Earth Aerocapture Trajectory, MTV



Hyperboloid 0.5 L/D Aerobrake





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Preliminary Results

Stagnation Point Heating (Park Method)

- Radiative heating for $30 < C_3 < 50\ km^2/sec^2$

  $\geq$ 80% of total (Park)

  $\geq$ 65% of total (Tauber-Sutton)

- Stagnation temperatures for $C_3 \geq 30\ km^2/sec^2$ exceed 2000° k

	Park		Tauber-Sutton	
C3	Q W/cm2	T °K	Q W/cm2	T °K
30	146.	2383	83.	2068
40	299.	2850	170.	2474
50	481.	3210	274.	2790
1993 technology ~	68.	1968		

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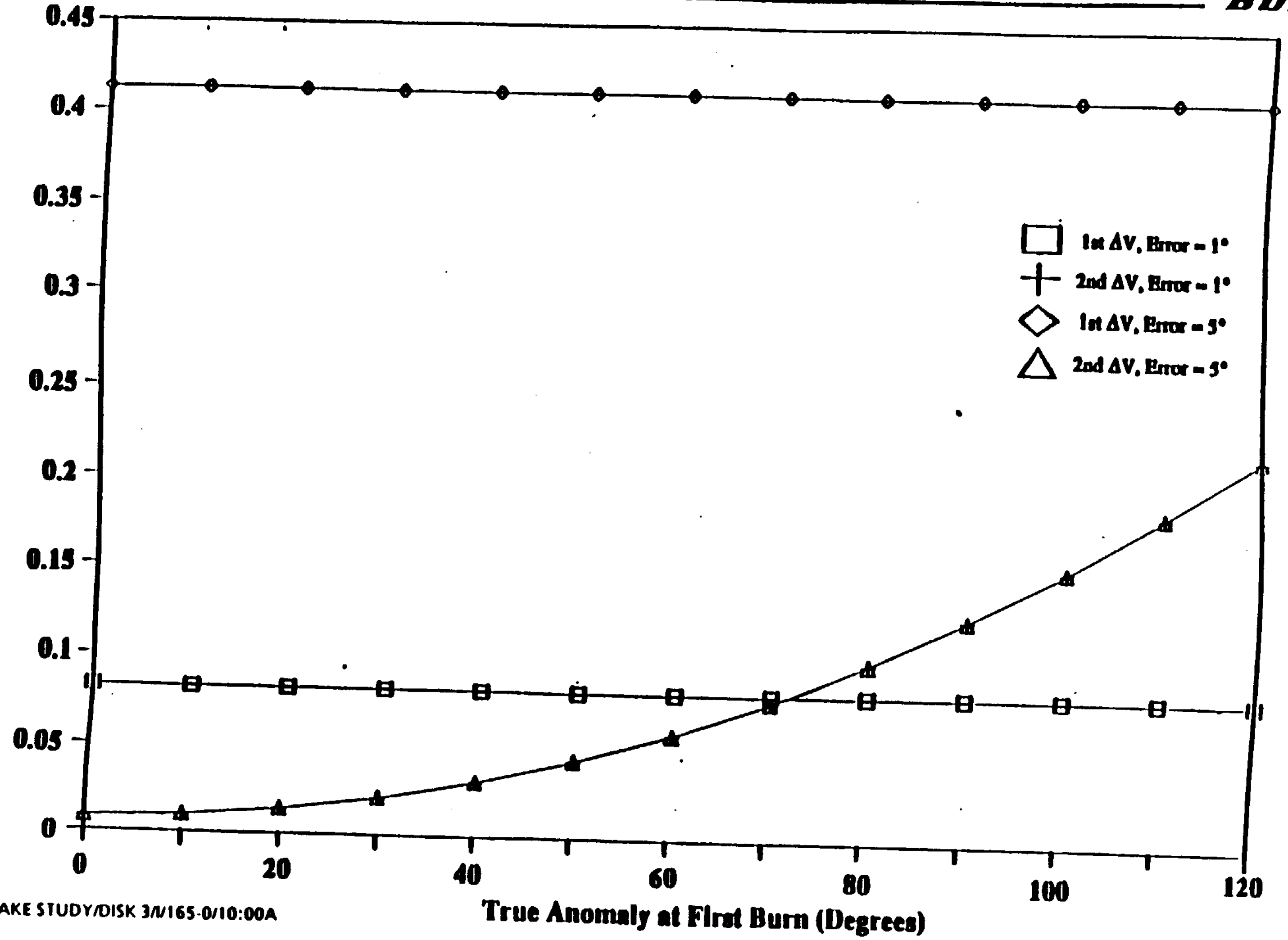
The delta velocity is independent on the true anomaly at first burn but is dependent on the flight path angle error. The two flight path angle errors examined were one degree and five degrees. Even though the delta V's for the different flight path angles are significantly different, the delta V for the second burn is not dependant on the error in the first burn. The total correction delta V is minimized by performing the first burn close to the periapsis where the true anomaly is zero.

Flight Path Angle Error on Delta V



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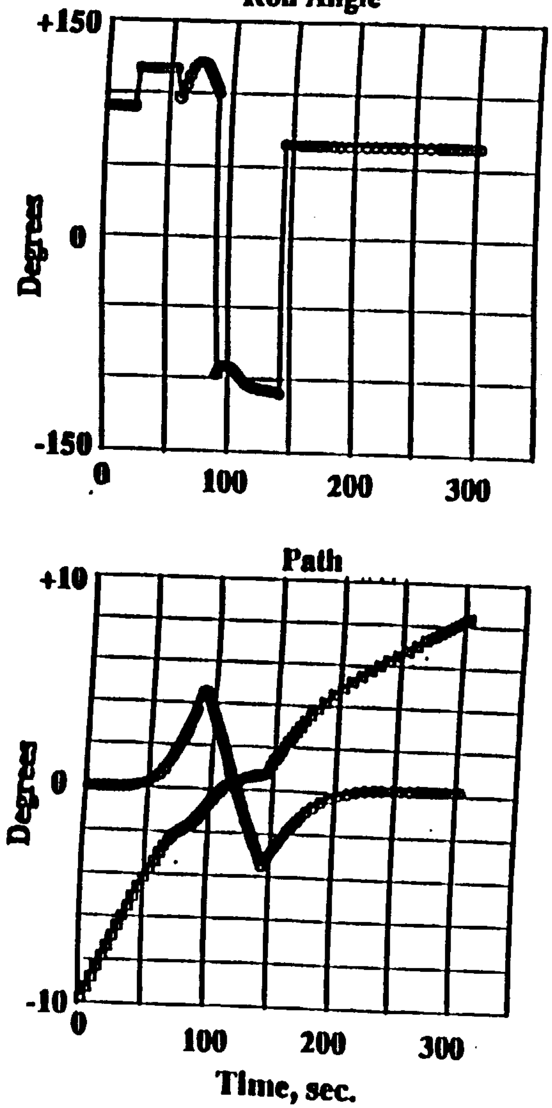
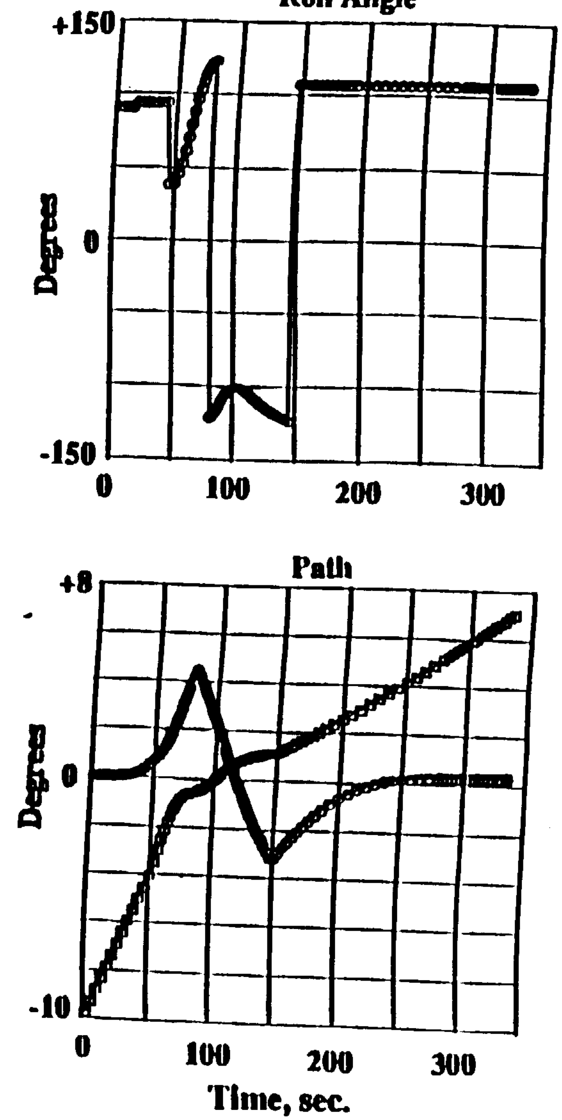
The delta velocity is independent on the true anomaly at first burn but is dependent on the flight path angle error. The two flight path angle errors examined were one degree and five degrees. Even though the delta V's for the different flight path angles are significantly different, the delta V for the second burn is not dependant on the error in the first burn. The total correction delta V is minimized by performing the first burn close to the periapsis where the true anomaly is zero.

Mars Aerocapture
Guided Trajectory Examples

COSPAR low-density atmosphere



COSPAR high-density atmosphere



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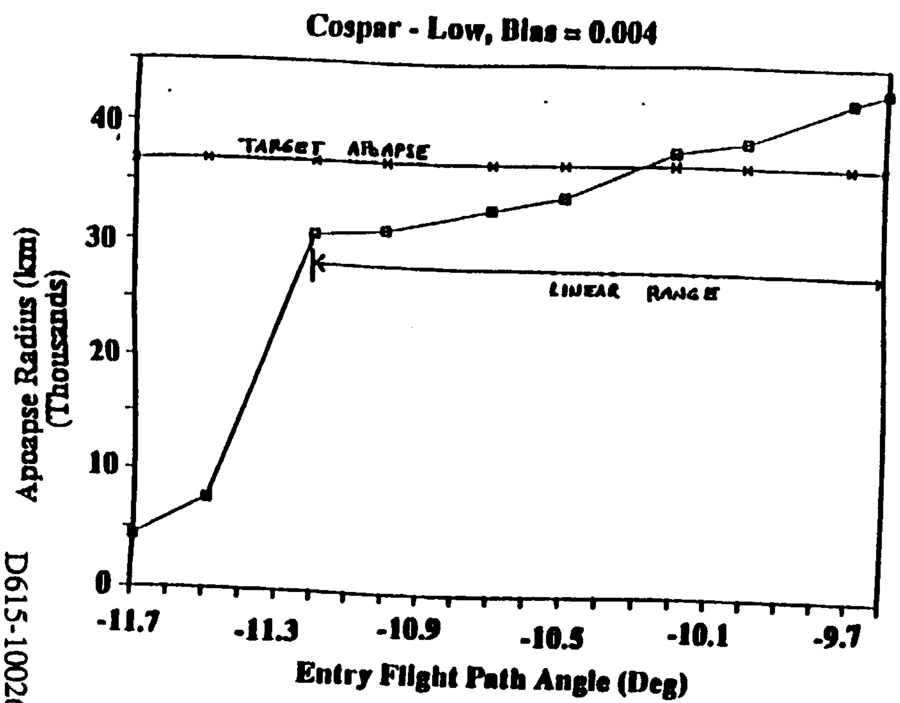
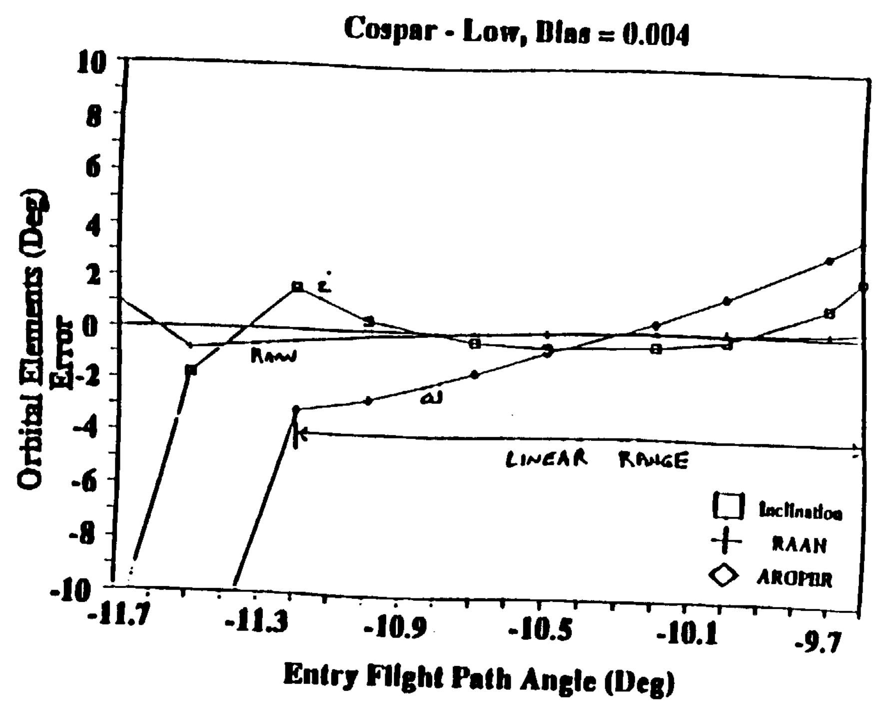
For the hyperboloid vehicle it is illustrated that the constraint can be met with an entry flight path angle of approximately ± 0.75°. This is illustrated for the COSPAR low atmosphere.

Parking Orbit Constraint Errors





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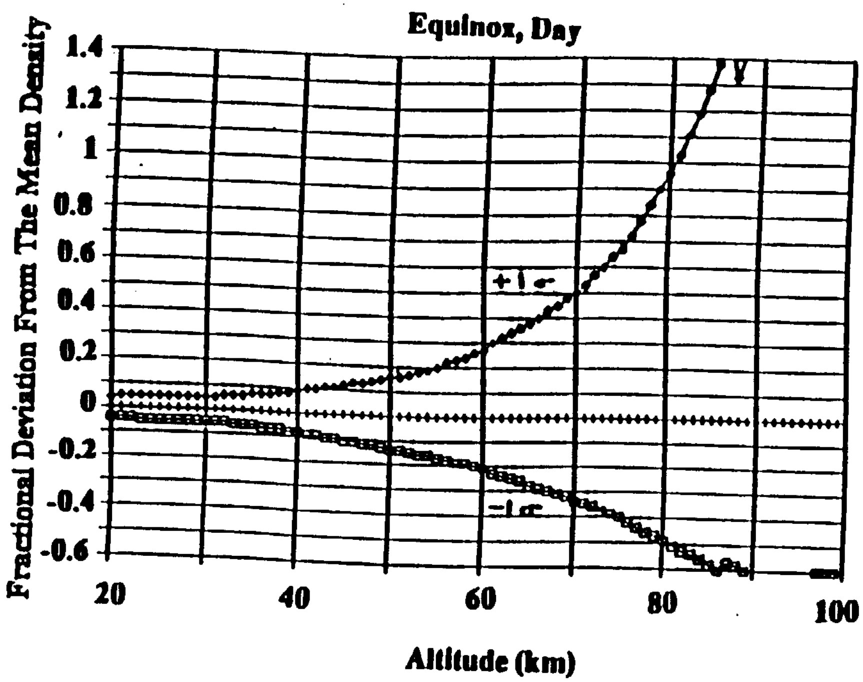
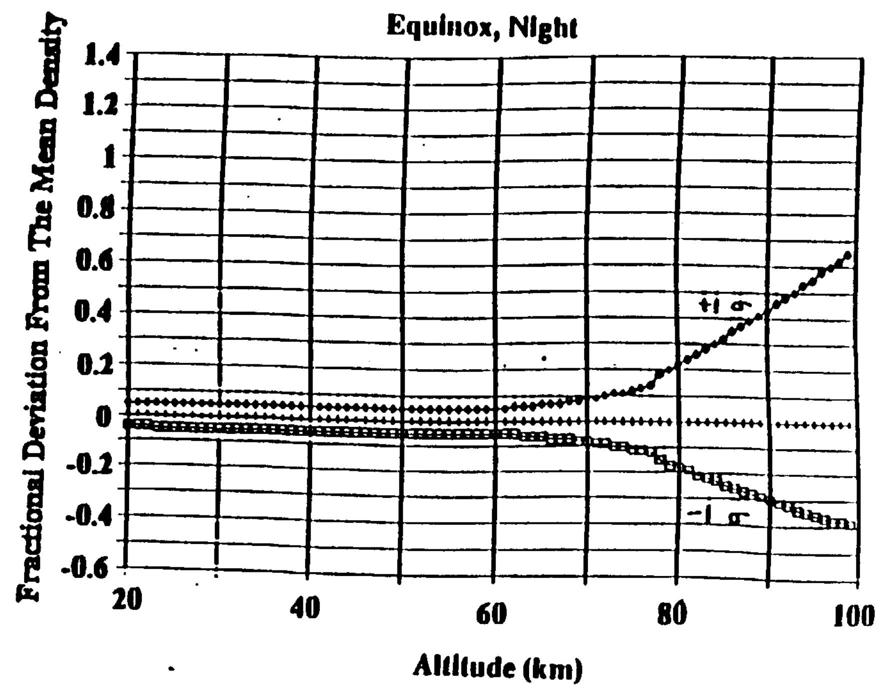
According to MarsGRAM, the one sigma density variation about the mean is illustrated for day and night at the equator during spring equinox of the year 2016. The wider envelope for the equinox day profile is due to the diurnal bulge.

Mars-GRAM Diurnal Variations of Density Envelope





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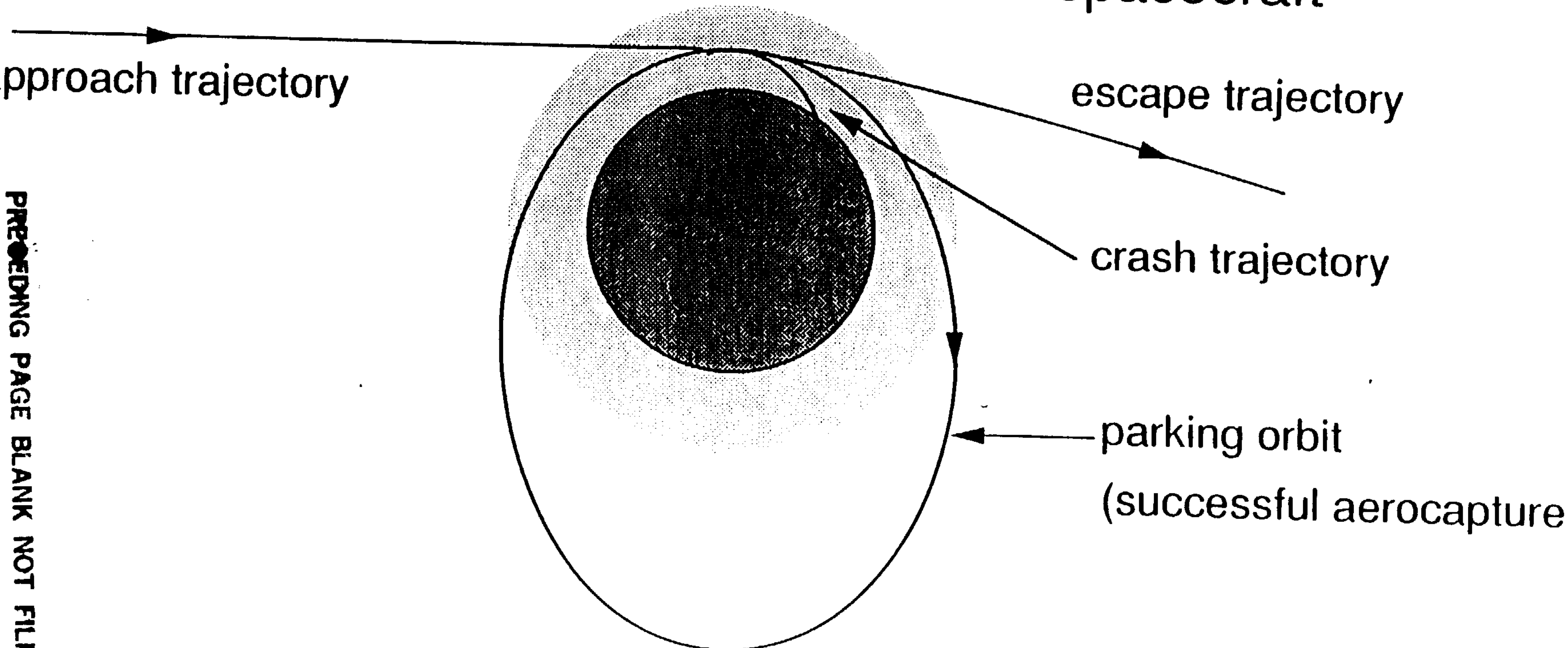
Guidance and Control at Mars

Problem

- Achieve successful aerocapture at Mars
  - Obtain desired parking orbit
  - minimize heating and g-loads

Technology thrust

- Apply artificial neural network technology to atmospheric prediction, and guidance and control of spacecraft



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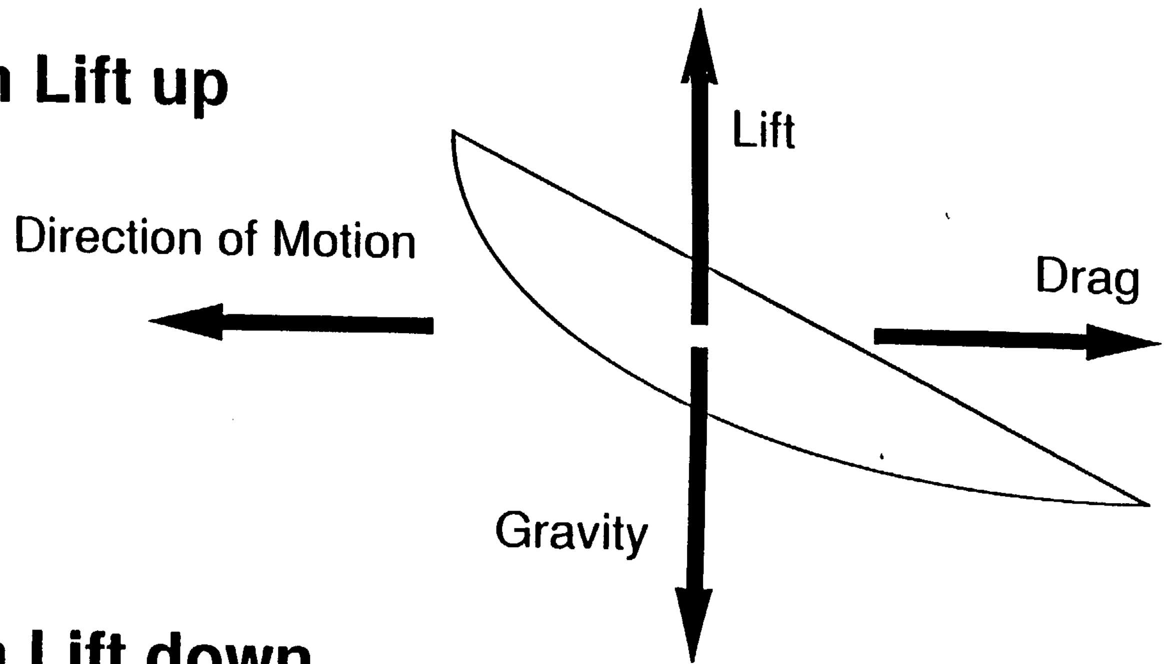
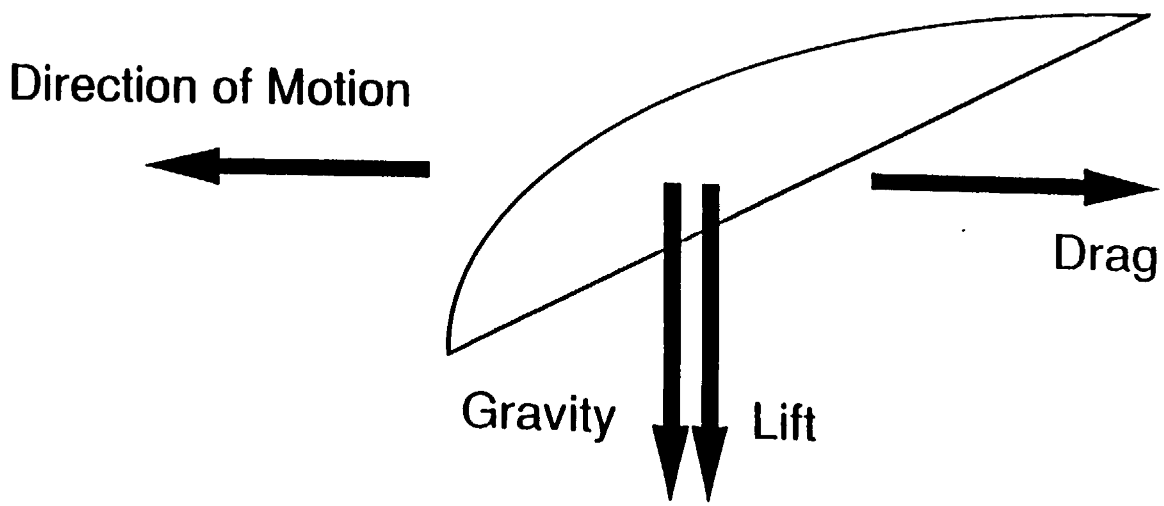
Aerocapture Corridor Definition

- **Aerocapture with Lift up**



- **Aerocapture with Lift down**



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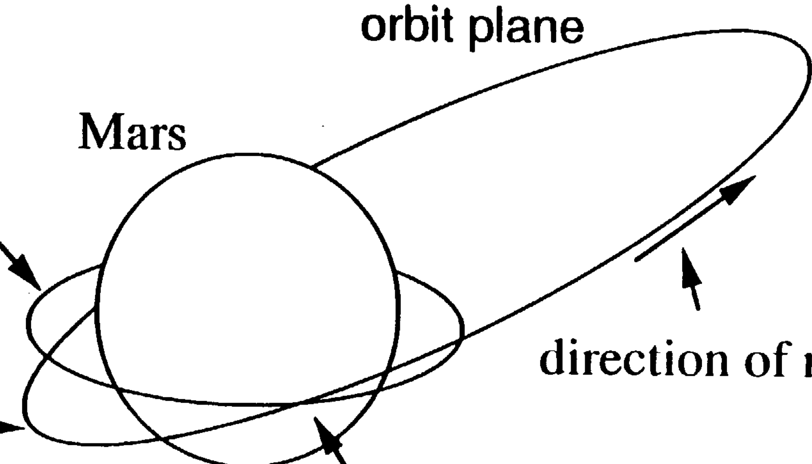
Guidance and Control Parameters

inclination = angle between the orbit and equatorial planes

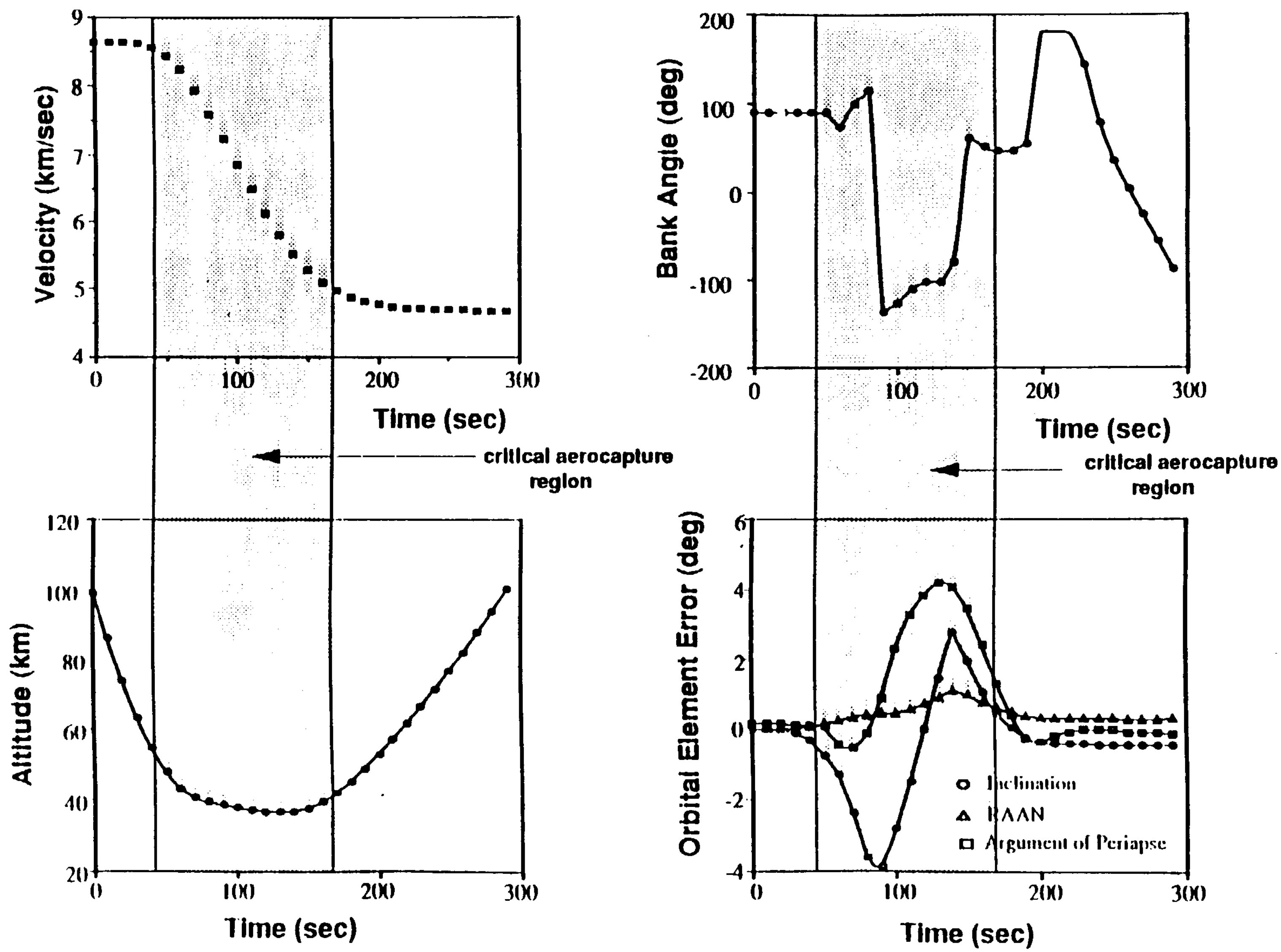


Guidance and Control Parameters

The left hand side of the chart shows the velocity depletion on aerocapture versus the altitude dip into the Mars atmosphere starting from 100 km altitude. The area shaded shows the critical area in which control authority must be exercised to effect the performance results of the aerobraked vehicle. To exercise this control, the vehicle enters the atmosphere with a bank angle of 90°. If the atmosphere is thinner than expected , the bank angle can be increased to increase the "negative lift" component and dive deeper into the atmosphere or if the atmosphere is thicker, to less steep bank can be used to compensate for the conditions and obtain an exit with the desired velocity.

The right hand side of the chart shows the new set of problems that are induced by the 90° bank angle. Mainly that the instantaneous orbit plane diverges from the target orbit plane. Therefore at least one bank reversal must occur to bring the two orbit planes back together.

Guidance and Control Parameters



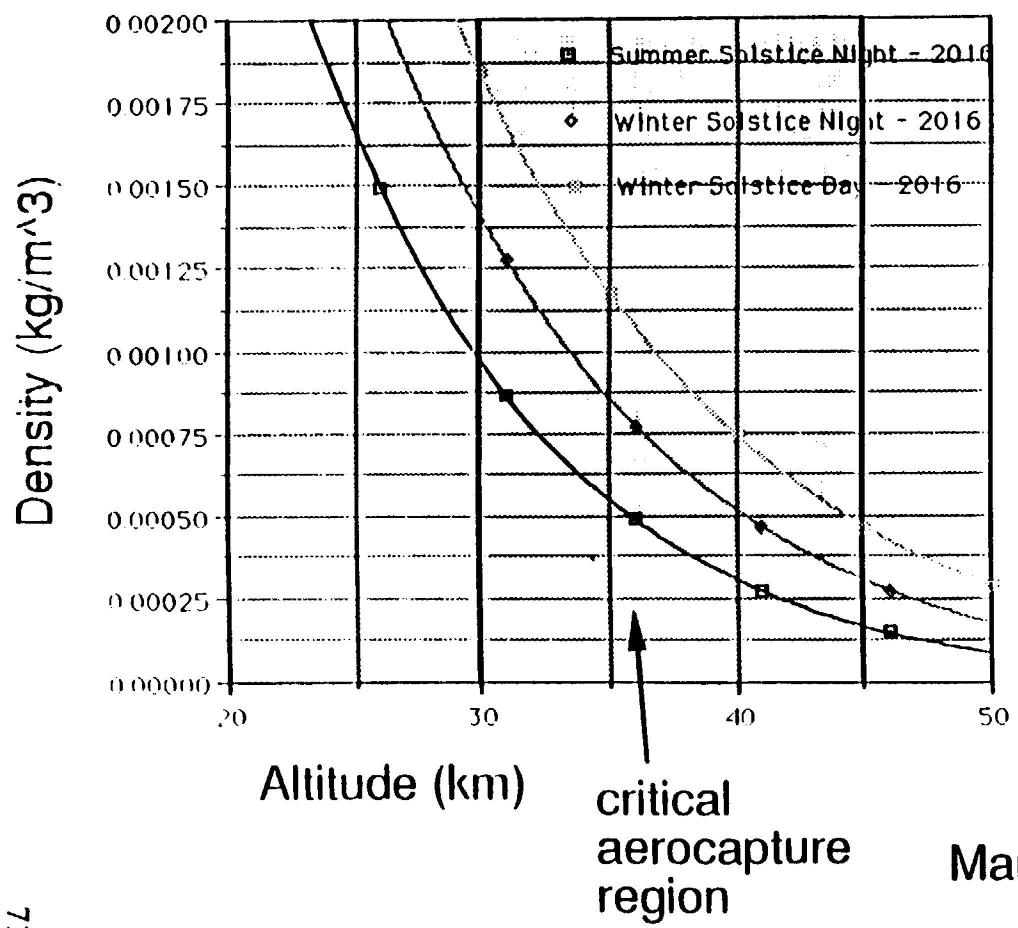
Guidance Disturbances and Errors

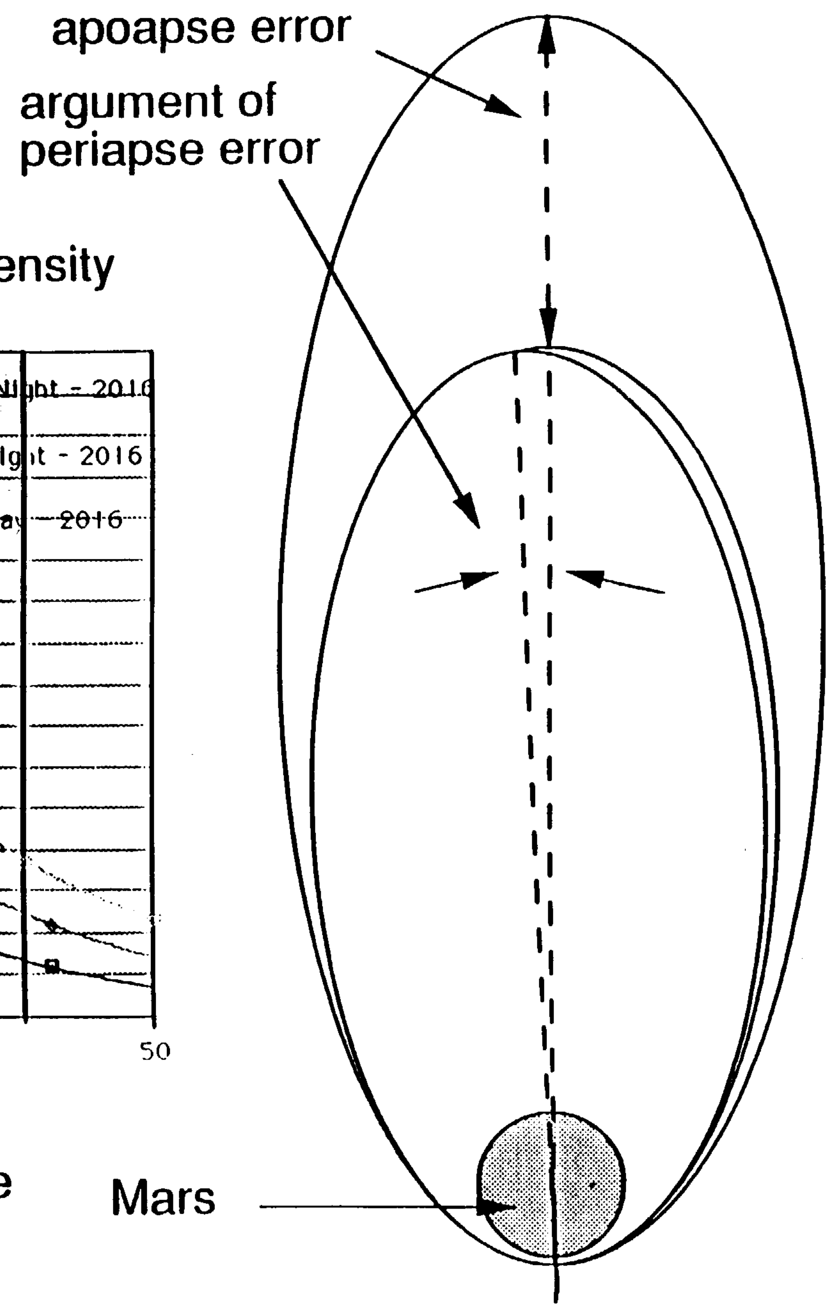
Mars density profiles play a major role in determining the bank angle profiles of the aerocapture trajectory. The plot on the left hand side of the chart shows the wide seasonal variation of equatorial density profiles as projected from known data. Note that the shaded part indicates the typical critical aerocapture region (i.e., most of the velocity depletion takes place in this region, therefore even minor density perturbations are a major concern).

The drawing on the right illustrates the effect of aerocapture on two parking parameters. Apoapse error is the most typical measurement. Much work has been done by Boeing and others on apoapse correction but, it is not the only parameter to be critically considered. For elliptical orbits, apsidal rotation is a major issue. Gross apoapse errors cost the same to correct as minute argument of periapse errors.

Guidance Disturbances and Errors

Mars Seasonal Equatorial Density Profiles for the Year 2016





Guidance Error Penalties

These plots illustrate the cost of the gross apoapse errors. A gross apoapse error of 15,000 km costs the same to correct as an apsidal rotation 1.5 degrees (60 m/s for a 1-Sol orbit). Even crude GN&C systems can constrain the apoapse to about +5,000 km. However none can control the apsidal rotation very well. Typical rotations are between 2 and 8 degrees.

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Guidance Error Penalties

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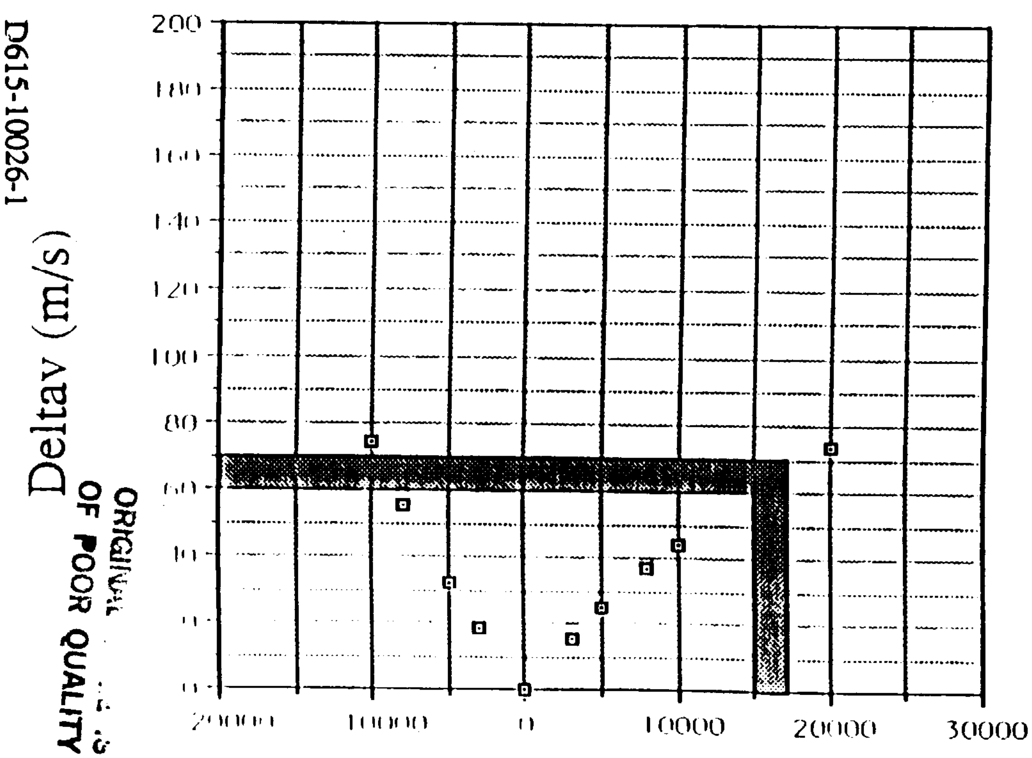
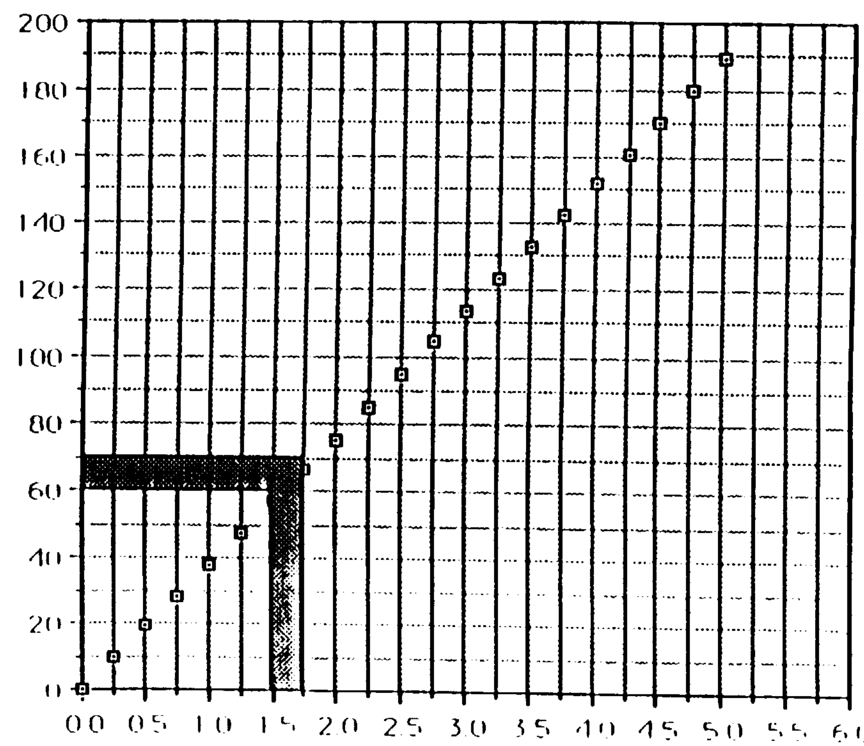
- 1 - sol orbit has a period equal to one solar Martian day -> 24.6597 hours

Guidance Error Penalties for 1 - Sol Orbit





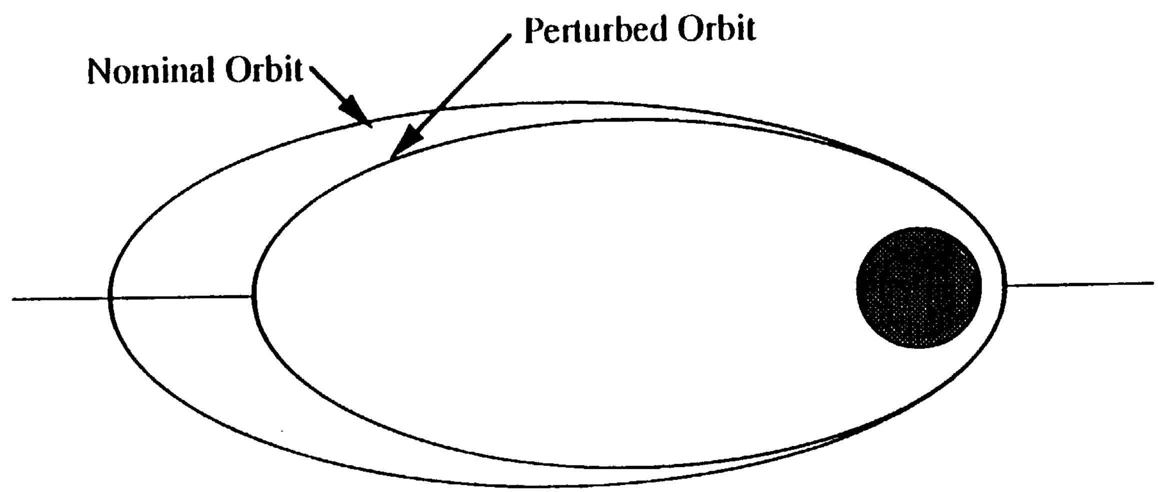
ORIGINAL OF POOR QUALITY

Parking Orbit Errors Due to Local and Global Dust Storm Effects

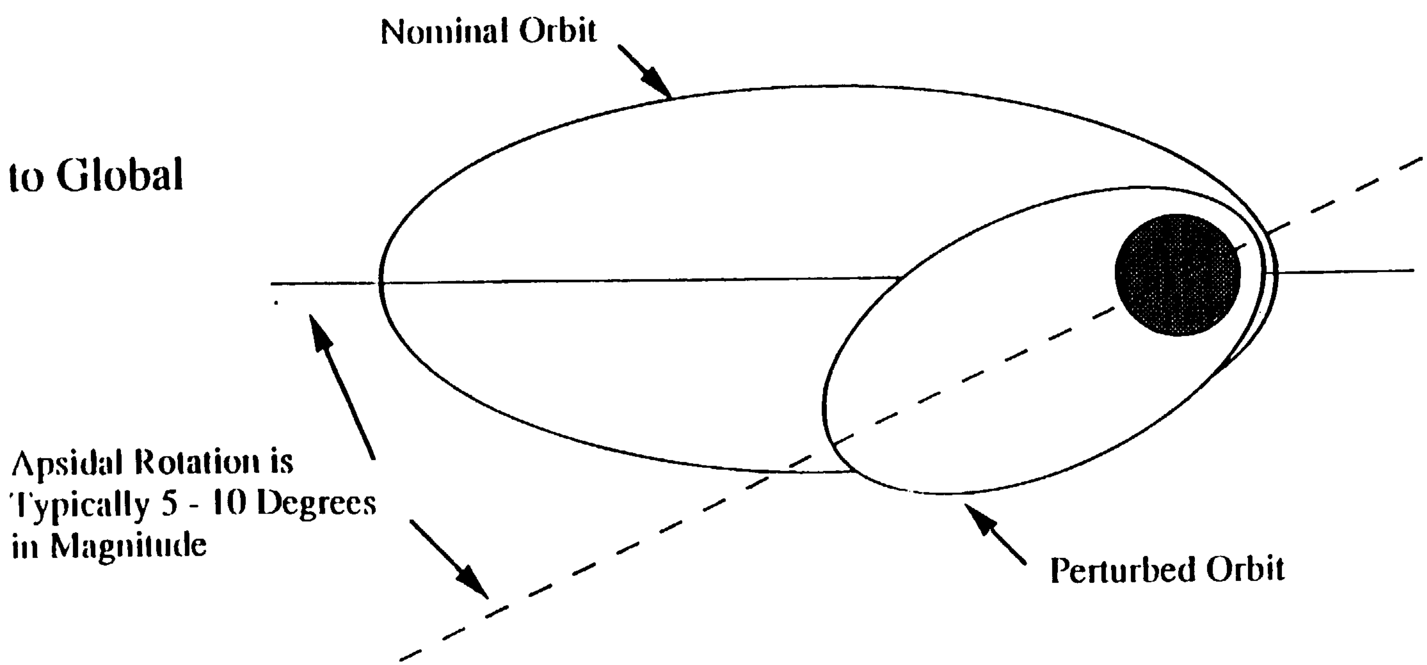
The effect of a dust storm on parking orbit errors depends on the type of storm. If it is localized, that is it occurs in a region below 40 km, then the apoapse errors are the most likely. If the storm is global, then apoapse and argument of periapse errors will probably occur. These are preliminary results.

Parking Orbit Errors Due to Local and Global Dust Storm Effects

Orbit Change Due to Local Dust Storm in the 30 - 40 km Altitude Range:



Orbit Change Due to Global Dust Storm:



A set of synthetic-density wave equations are given on the following These wave equations were used to determine aerocapture guidance sensitivities from density variations (worst case) that may be encountered during Mars Aerocapture.

Synthetic Density Wave Equations

- **Horizontal sine-wave density scaling**

  DENS = DENS*[1.0 + 0.3*SIN((Y/W)*2Pi)]

- **Horizontal sine-wave density and vertical density ratio scaling**

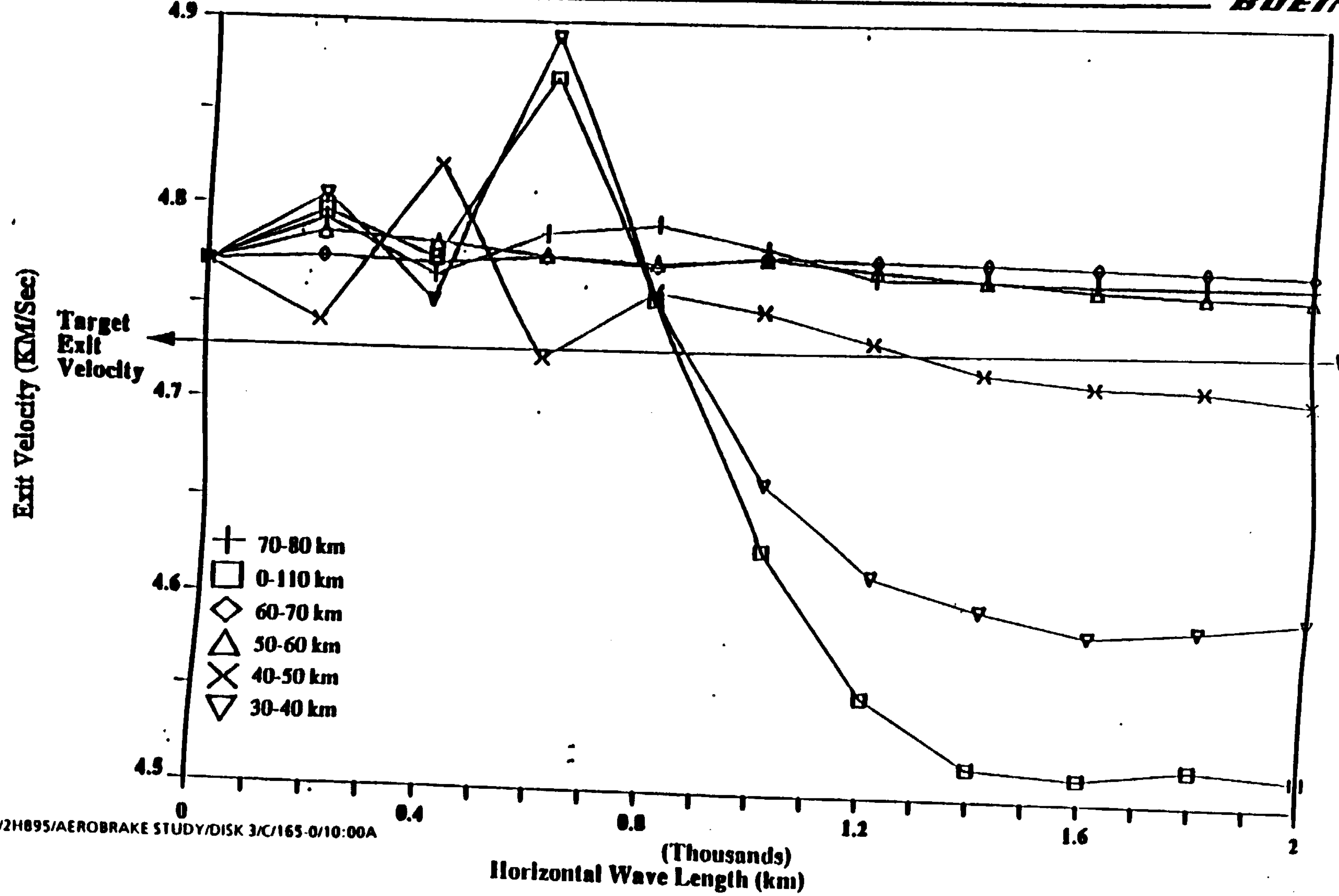
  DENS = DENS*[1.0 + 0.05*SQRT (DENS30/DENS)*SIN((Y/W)*2Pi)]

- **Horizontal sine-wave density and vertical density-ratio and sine-wave density scaling**

  DENS = DENS*[1.0 + 0.05*SQRT (DENS30/DENS)*SIN((Y/W + H/LZ)*2Pi)]

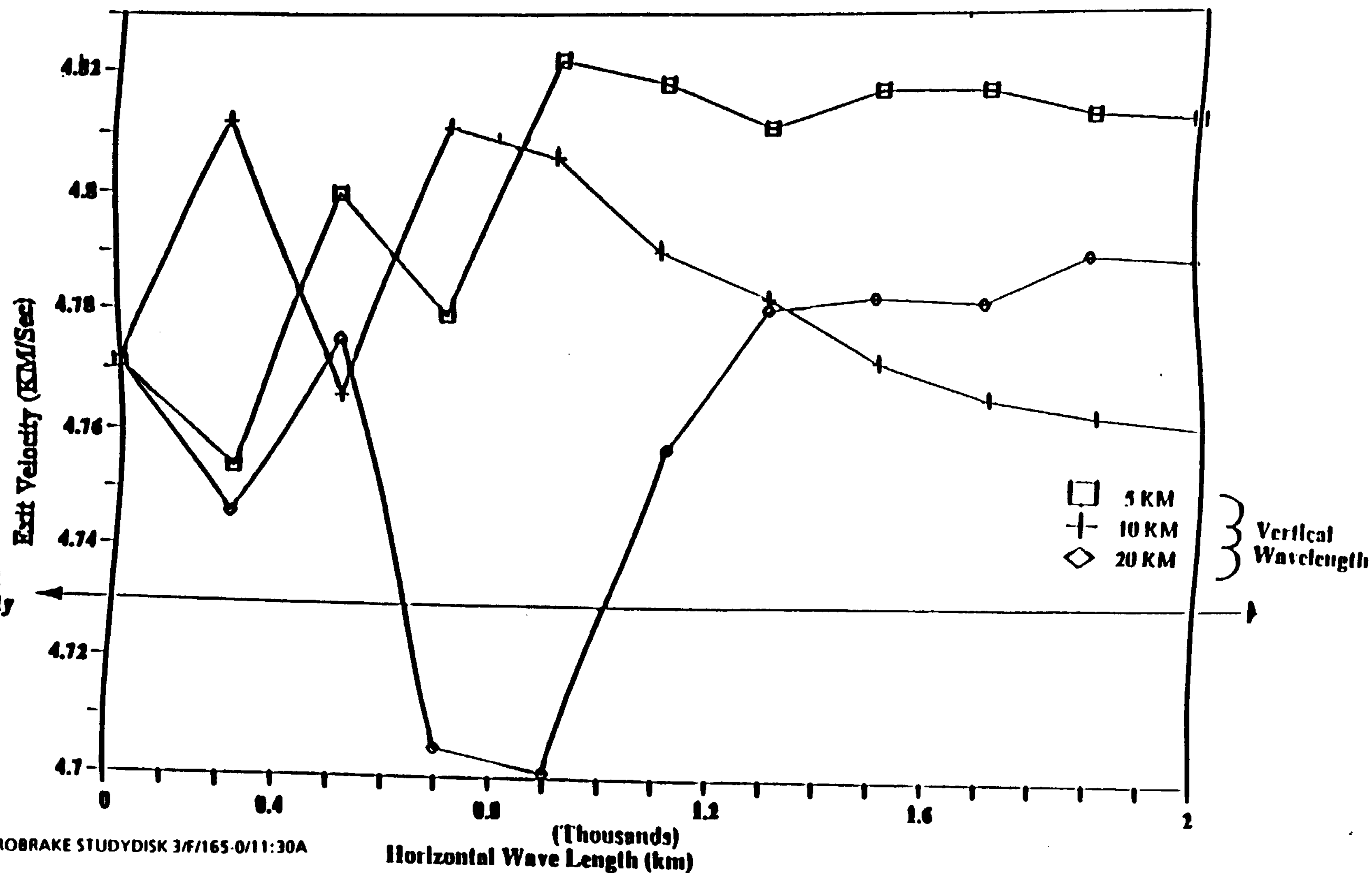
Using the COSPAR low atmosphere with sine wave distribution of the density, calculations were made which illustrate that for sine wave lenghts greater than 1000km the exit velocity errors are higher than for the lower wave lengths. The 30 - 40 km altitude region is by itself the most critical region.

Horizontal Sine Wave
Density Scaling



Guided trajectories were simulated for horizontal and vertical wave lengths. The horizontal wave length varied to 2000 km with vertical wave lengths of 5, 10, and 20 km. The larger vertical wave length of 20 km provides a more favorable atmospheric condition as the density variation in the vertical direction does not vary as much as the lower wave lengths in the critical region.

Horizontal Sine Wave Density and Vertical Density Ratio and Sine Wave Density Scaling



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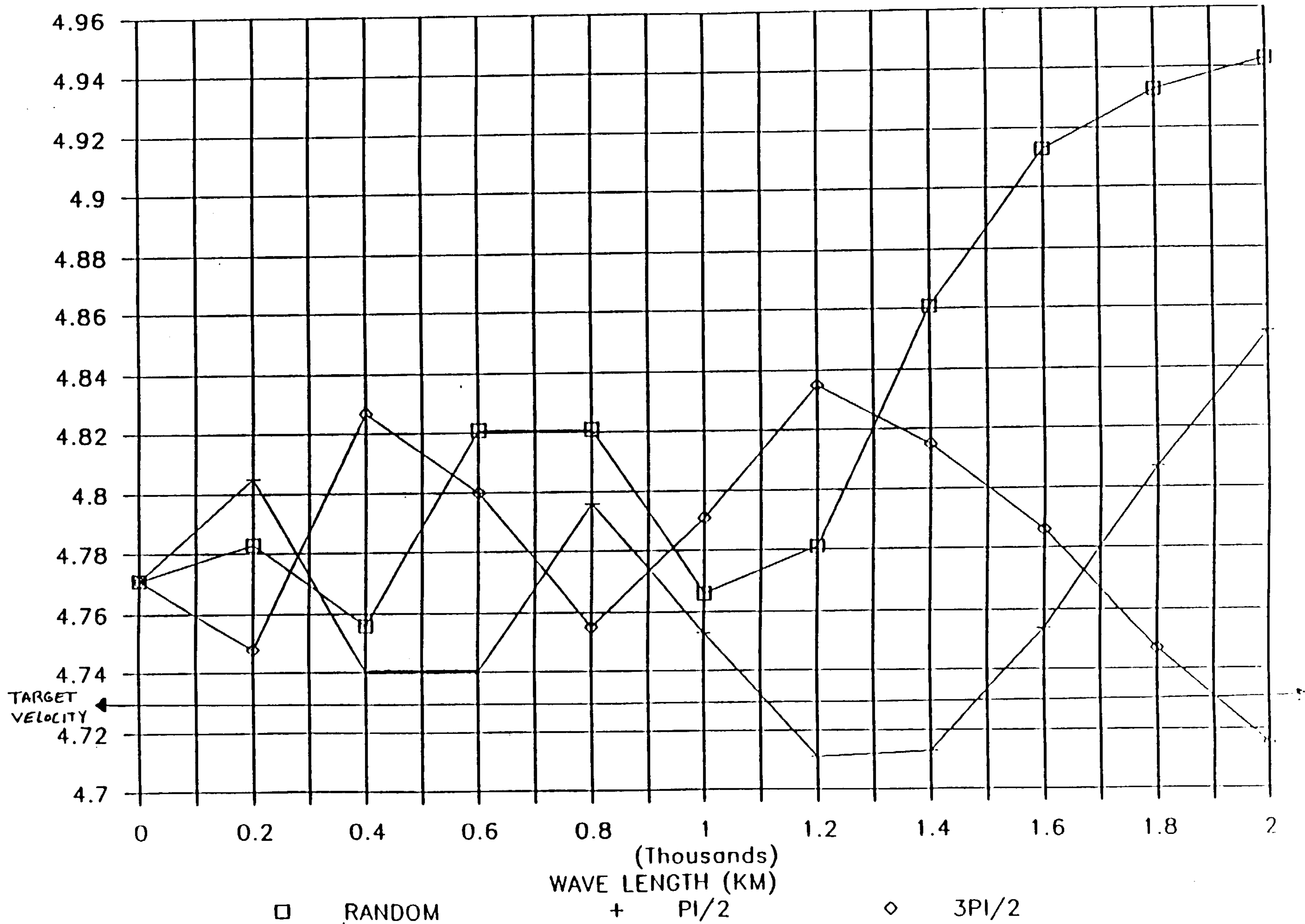
HORIZONTAL SINE-WAVE DENSITY AND VERTICAL DENSITY RATIO SCALING WITH SINE CURVE PHASE SHIFT



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Adaptive and Non-Adaptive Guidance

This chart illustrates the effect of using atmosphere prediction early in the trajectory. The equation at the top is the density perturbation equation used to simulate disturbances in the Martian Atmosphere.

Alpha (α)is the amplitude of the density wave, lambda (Λ) is the density wave wavelength, and theta (Θ) is the phase angle of the wave. For this analysis the amplitude was was held to 0.10 or 10%, the wavelength was varied between 0 and 3000 km and the phase was set at 45 degrees.

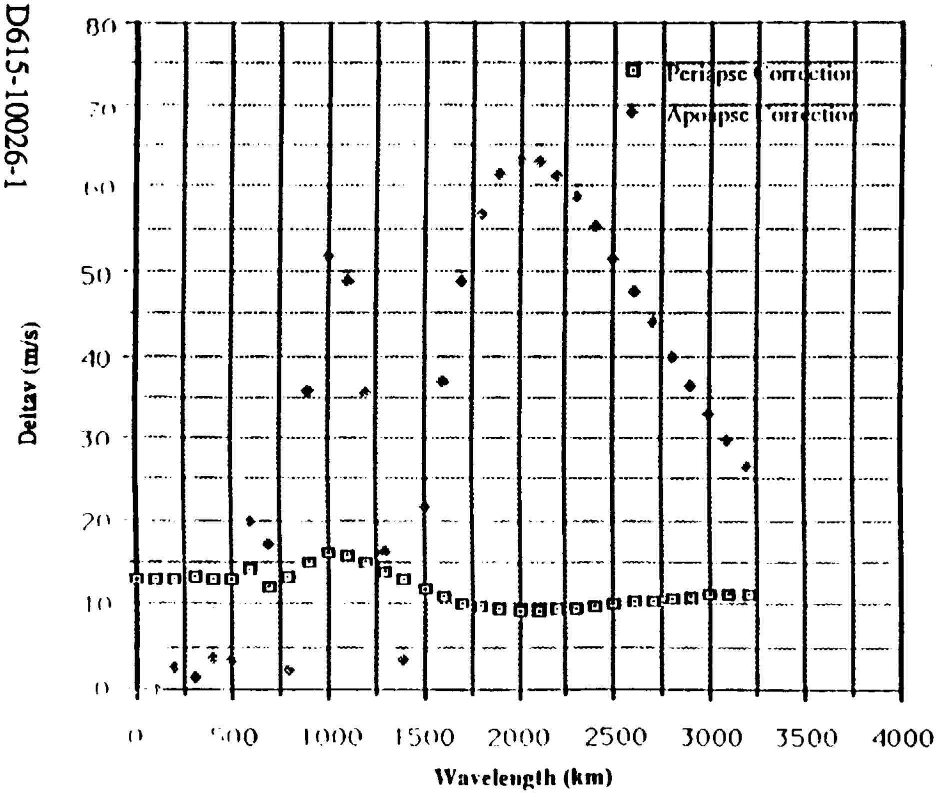
The first plot shows the effect of wavelength on the apoapse ΔV penalty. Each pair of apoapse/periapes ΔV points correspond to a guided trajectory.

The second, right hand, plot shows the effect of using a "wave predictor" for the first 60 seconds of the trajectory. The GN&C samples density data for this short interval of time and them estimates the wavelength. A predetermined gain is interpolated from a table which biases the flight profile so that the end constraints can be met. Although these results are preliminary, they show that atmospheric prediction/ characterization can enhance GN&C performance.

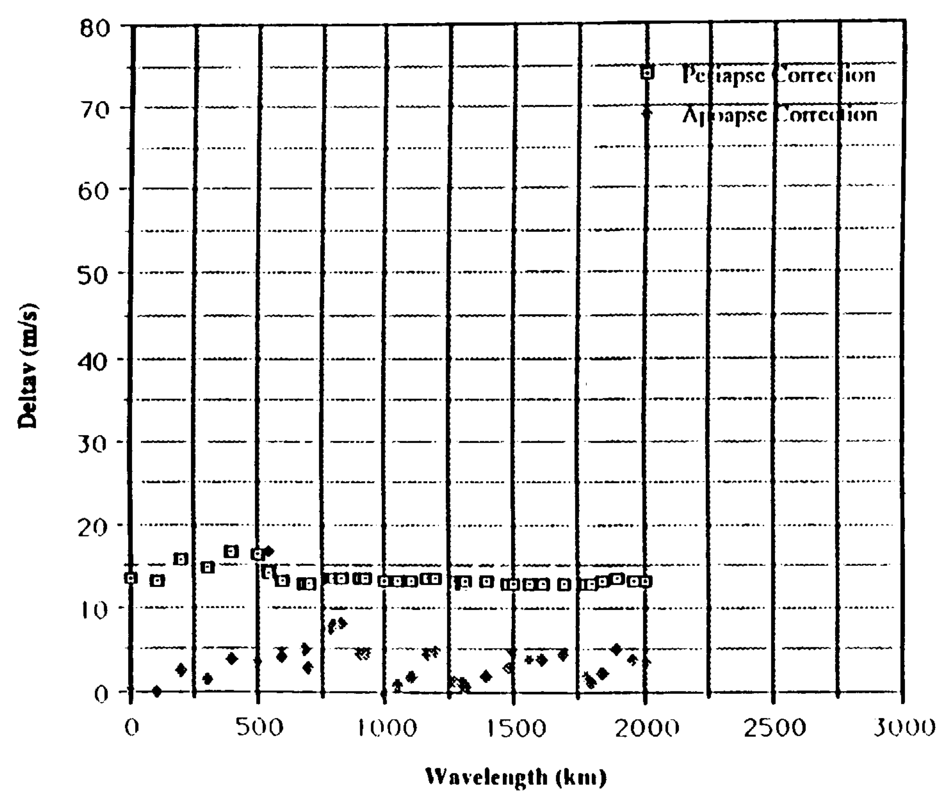
Adaptive and Non-Adaptive Guidance

$$\rho = \rho*[1.0 + \alpha*\mathrm{SIN}((2\Pi/\Lambda) + \Theta)]$$

Periapse and Apoapse Correction Deltavs for 10% Density Wave Amplitude: Non-Adaptive Guidance



Periapse and Apoapse Correction Deltavs for 10% Density Sine Wave Amplitude: Adaptive Guidance



Standard Orbit Analysis

This analysis was done to evaluate the feasibility of leaving a module in orbit at Mars and reencountering it on a succeeding mission without a large ΔV penalty. Both the Level II and Boeing Nominal missions were used as the initial mission in which an MEV was left in orbit. The standard orbit was propagated according to which mission was being studied. The succeeding missions were the 2018, 2022 and 2024 missions. Orbit inclination and period were varied for all missions to find a compromise "fit". At present we have not found a standard orbit that is adaptable to both the original and return missions without insertion or departure ΔVs of from 2 to 6 km/sec.

Standard Orbit Analysis

Asssumptions

- High thrust missions
- MEV remains in orbit about Mars
- May perform small period trims

Method

- Use 2016 as base mission
- Use 2018, 2022, and 2024 as return missions
- Vary inclination and period

Results

- No orbit in which both insertion and departure deltav are small
- Will have to customize initial and return missions to use standard orbit
- More work is required for a complete understanding of the standard orbit

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Two structural designs, spar and truss, were used for the hyperboloid aerobrake. The aerobrake has a length of approximately 31 meters, width of 28.2 meters and height of 6.5 meters. In both cases the aerobrake accommodates an engine hatch. In the case of the truss configuration, the tetrahedral truss respective points were projected on to the aerobrake surface to accommodate the required curvature. The spar configuration used a carbon magnesium metal matrix and the truss configuration used graphite epoxy. In both cases an aluminum honeycomb core was employed with a titanium face sheet. The Mars excursion vehicle had a payload of 81 metric tons and the Mars transfer vehicle the payload was 153 metric tons.

Aerobrake Structural Design

Design Assumptions:

- Constant spar cross-sections, curved profiles
- C/Mg metal matrix spars (density 1830 kg/cu. m.)
- Payload: Mars Excursion Vehicle, 81MT
- 6g maximum acceleration
- 8 payload attach points (4 frame and 4 landing leg points)
- Relative wind angle = 20 degrees
- Variable pressure distribution, range 1.5psi to 3psi
- Structure temperature = 394K (250F)
- Secondary spar pattern to be triangular for greater shear resistance

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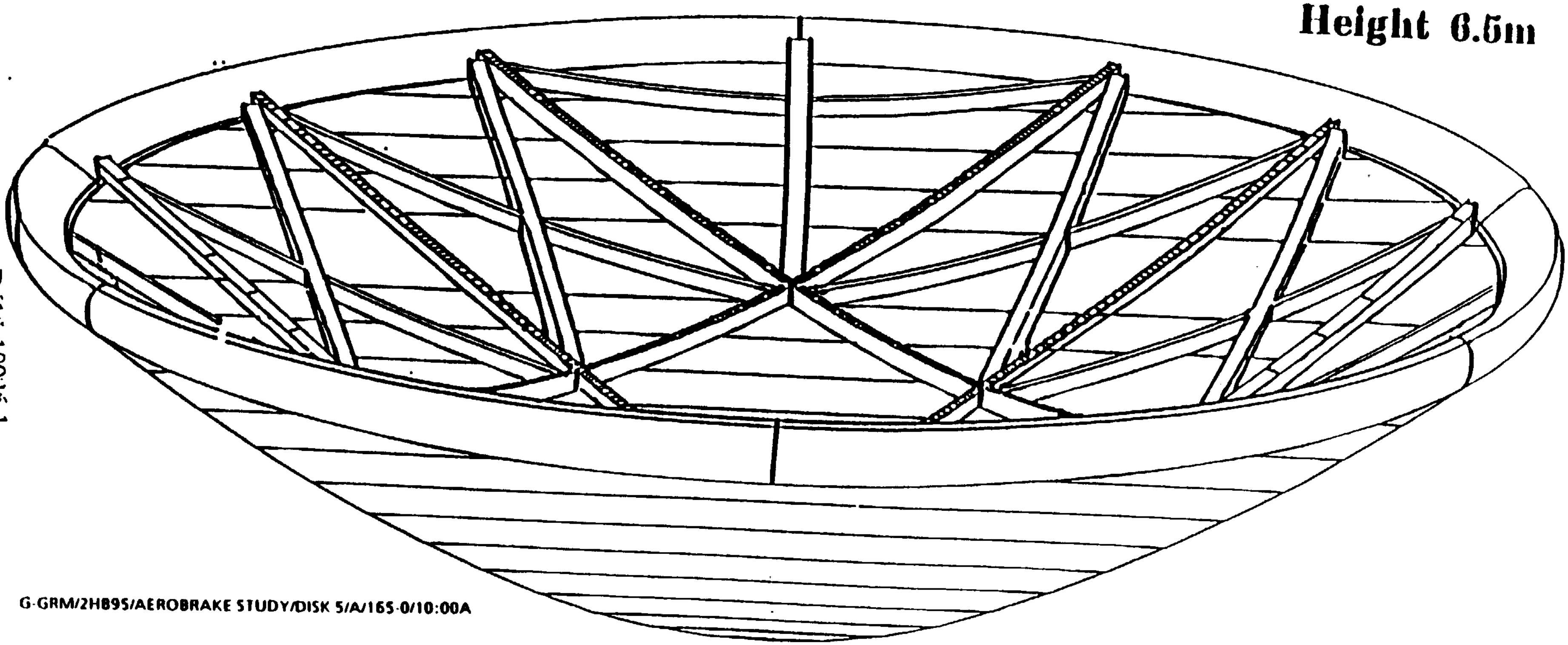
Structural Concept - Spar Configuration
3D View

Size

Length 31.1m

Width 28.2m

Height 6.5m



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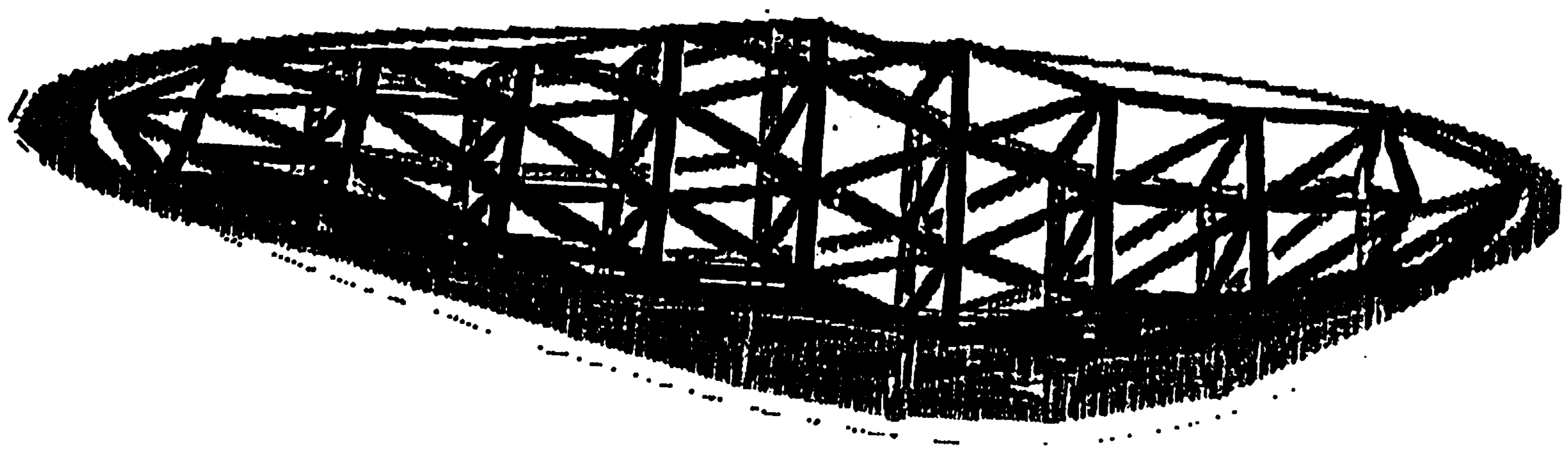
Structural Concept - Truss Configuration

3D View



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Aerobrake Structural Design
81 mt payload, MEV

19.5 inch spar depths:	105 ksi spar strength		200 ksi spar strength	
Primary spar weight:	5,390 kg	(11,859 lb)	2,571 kg	(6,052 lb)
Secondary spar wt:	3,827 kg	(8,420 lb)	2,975 kg	(6,546 lb)
Honeycomb weight:	6,758 kg	(14,868 lb)	6,758 kg	(14,868 lb)
TPS weight:	3,300 kg	(7,260 lb)	3,300 kg	(7,260 lb)
Total aerobrake weight:	19,275 kg	(42,407 lb)	15,784 kg	(34,726 lb)
22.5 inch spar depth:				
Primary spar weight:	4,989 kg	(10,978 lb)	2,484 kg	(5,465 lb)
Secondary spar wt:	3,809 kg	(8,379 lb)	2,596 kg	(5,711 lb)
Honeycomb weight:	6,758 kg	(14,868 lb)	6,758 kg	(14,868 lb)
TPS weight:	3,300 kg	(7,260 lb)	3,300 kg	(7,260 lb)
Total aerobrake weight:	18,856 kg	(41,483 lb)	15,138 kg	(34,726 lb)

Note: 200 ksi option may require additional material technology development efforts.

G-GRM/2H895/AEROBRAKE STUDY/DISK 4/L/165-0/11:00A

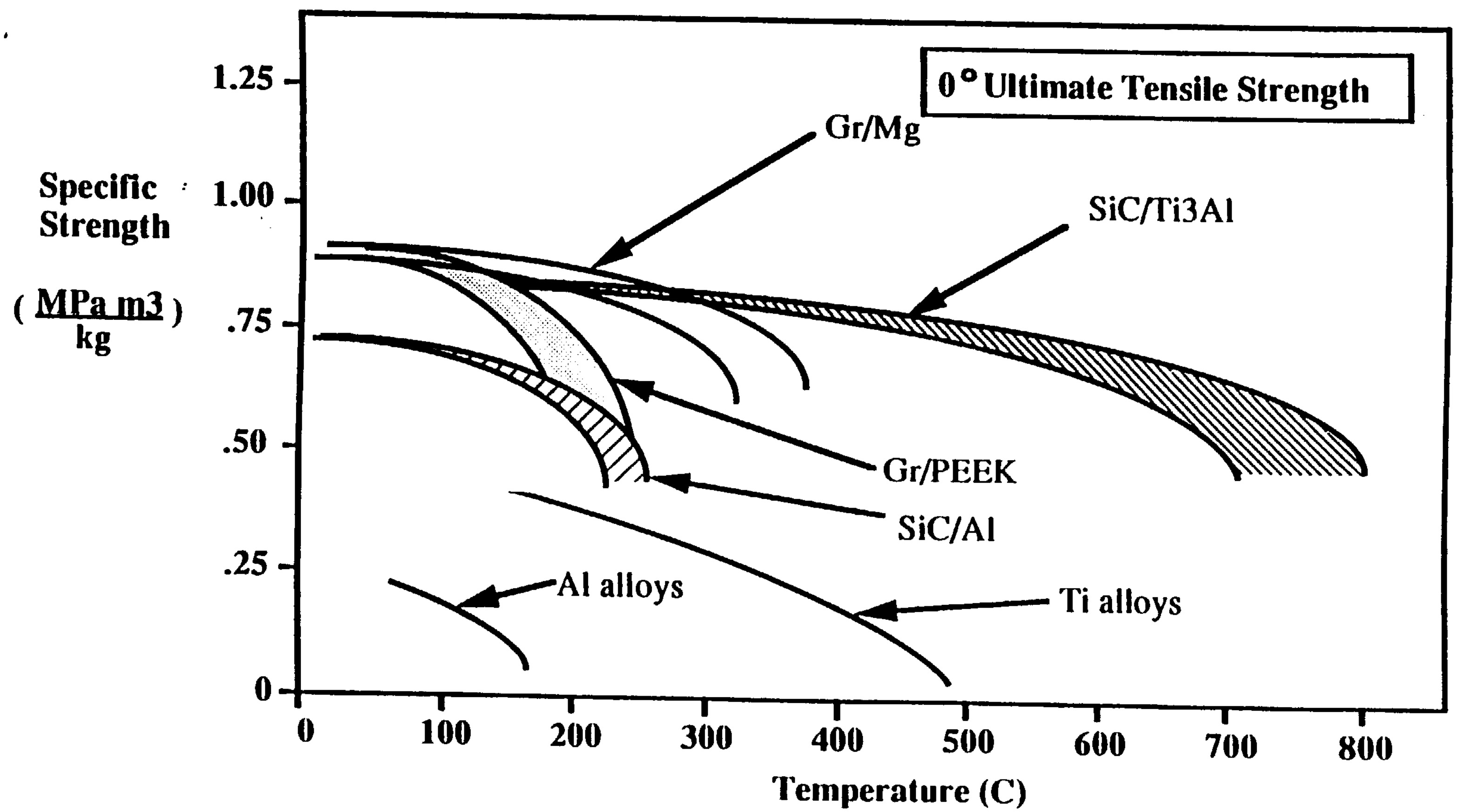
Specific Strength of Materials

Materials having high strength-to-weight and stiffness-to-weight ratios were selected to minimize structural mass.

Projected trends in new structural materials are summarized on the facing page.

Gr/Mg shows the highest strength-to-weight ratio at temperatures less than 300 C and was chosen for internal structural frames, longerons, and truss members. Because of its superior strength at high temperatures, silicon carbide reinforced titanium aluminide (e.g. alloy Ti-14Al-22Nb) was chosen for the heat shield, stabilizers, and elevons. About 30% of the lower surface area requires a thermal barrier coating to prevent excessive heating. Silicon carbide reinforced aluminum was the favored material for upper surface skins where payoff for high strength is secondary to durability and environmental considerations.

Specific Strength of Materials



High L/D Aerobrake Temperature Distribution

The figure on the facing page presents the temperature distribution which was used for performing the materials and structural analyses. The temperatures are based on computations of the boundary layer temperatures around the aerobrake during a 2g descent.

A primary consideration in the selection of structural materials is the maximum operating temperature and the total heat load transmitted to the structure during descent.

ADVANCED CIVIL SPACE SYSTEMS

High L/D Aerobrake Temperature Distribution

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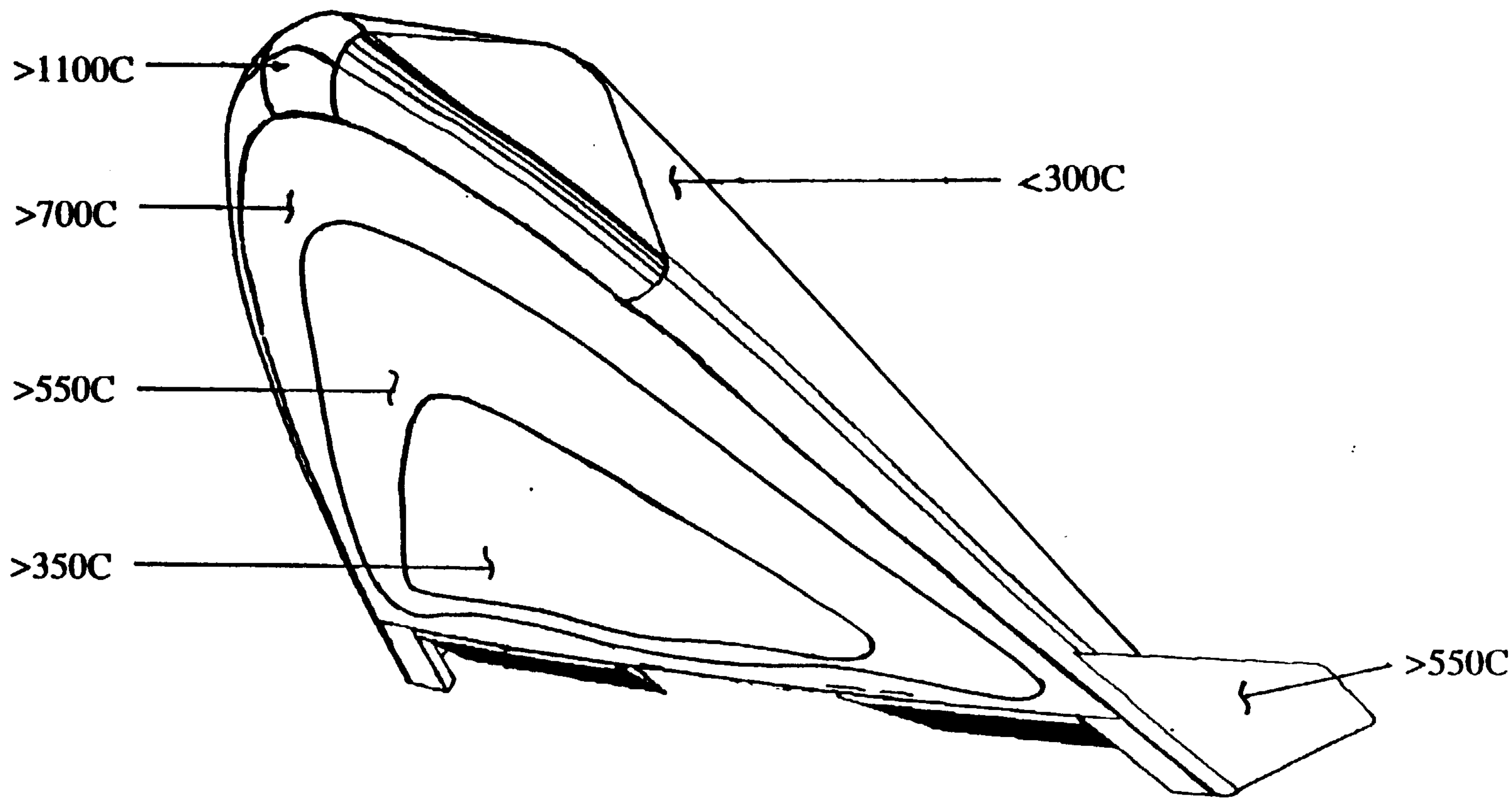


Preliminary Structural Design Assumptions

Aerobrake contract/highldtemp/bew/21Sep90

Aerobrake Materials Assumptions

Materials selections for the high l/d aerobrake are based on forecasts of future trends in materials technologies. Structural properties were derived by combining or extrapolating existing data. For example, information about the behavior of composites of silicon carbide and conventional titanium alloys was combined with data on the effects of high temperature on titanium aluminides. In the prediction of composite properties for fiber volumes exceeding available data, the "rule of mixtures" was assumed.

Properties of superplastically formed/laser welded "egg crate" core are derived from Boeing R&D sources and properties of equivalent density titanium honeycomb. "Egg crate" core structures have been superplastically formed at a density of 480 kg/m3 using conventional titanium alloys, but achieving a core density of 160 kg/m3 with titanimu aluminide represents a technology development need.

Hot Structure Materials Assumptions for High L/D Aerobrake

Structural Application	Materials	Maximum Operating Temp.	Strength at Temperature Approximations	Density (kg/m3)
Thermal barrier (over 30% of surface which exceeds 700 C)	• Plasma sprayed zirconia over felt metal SIP	700+ C	Fcy = 90 MPa Ftu = 125 MPa Ec = 2.6 GPa	3128
Structural heat shield (lower surface), tail, elevons, and engine hatch doors	• skin: 55% SiC reinforced Ti3Al	700 C	Fcy = 1897 MPa Ftu = 1965 MPa Ec = 183 GPa	3775
	• core: "egg crate" Ti3Al (SPF/LW)	550 C	Fsw= 4.76 MPa Fsl = 6.86 MPa Gw = 690 MPa Gl = 745 MPa	160
Structural frames, longerons, & truss memebers	• 45% graphite reinforced Mg alloy	350 C	Fcy = 1241 MPa Ftu = 1379 MPa Ec = 193 GPa	1860
Structrual skins, upper surface	• 45% SiC reinforced Al alloy	<300 C	Fcy = 1531 MPa Ftu = 1586 MPa Ec = 207 GPa	2609

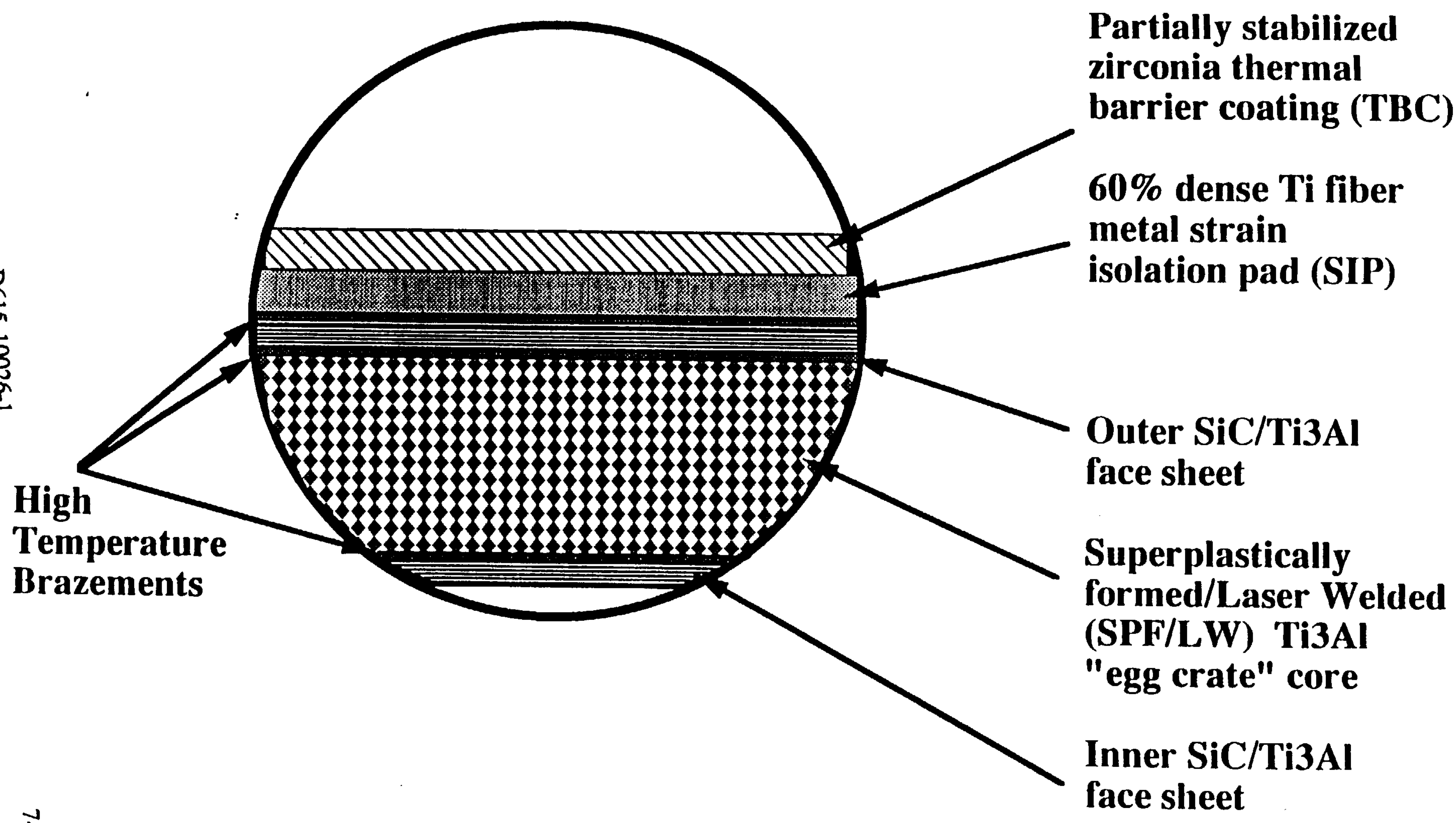
Aerobrake Shell Materials

The figure below represents a cross section view of the aerobrake heat shield.

The heat shield utilizes a thermal barrier coating in the high temperatures in the regions. The maximum operating temperature of the Ti3Al composite sturcture is limited to 700 C. A strain isolation pad is necessary to accommodate the differences in thermal expansion between the coating and face sheet. Silicon carbide fibers in the face sheets serve to channel the heat away from the high temperature regions.

The top surface of the aerobrake is a skin and stringer construction of SiC/Al reinforced by the C/Mg frame members. The bottom surface is also reinforced with C/Mg frame members (not shown).

Hot Structure Materials Assumptions for High L/D Aerobrake



High L/D Aerobrake Pressure Distribution

The figure below shows a representative pressure distribution which was assumed in performing the structural analysis. The pressures were used in approximating loads on the shell panels and the frame members.

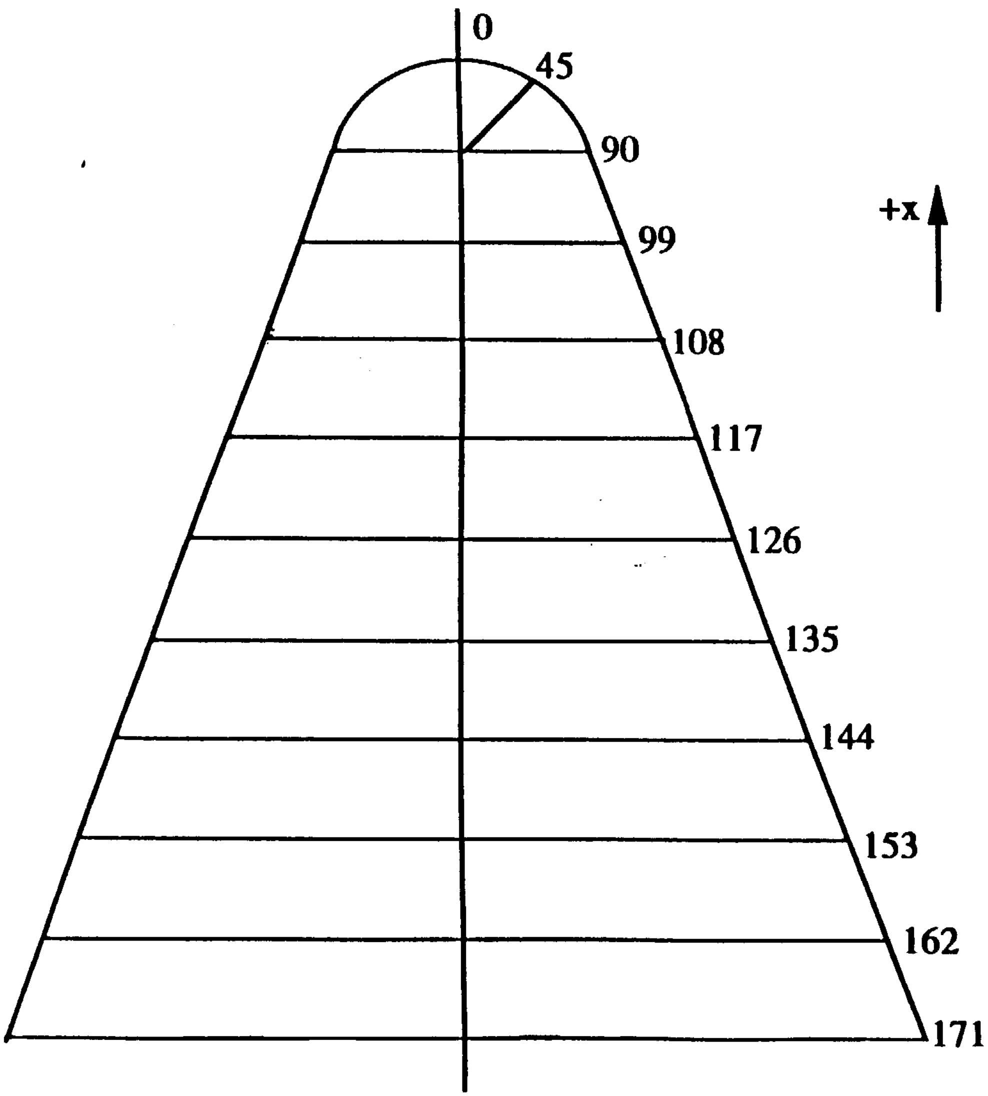
The maximum pressure at the lip is shown along with the minimum pressure at the corresponding reference stations. The minimum pressure was approximately uniform away from the lip of the aerobrake.

Note that the pressure distribution decreases toward the rear of the aerobrake, but the panels become larger. This structural trade-off allowed the use of common panel cross-sectional dimensions without a weight penalty.

High L/D Aerobrake Pressure Distribution

(3)

Descent (2g max. acceleration)



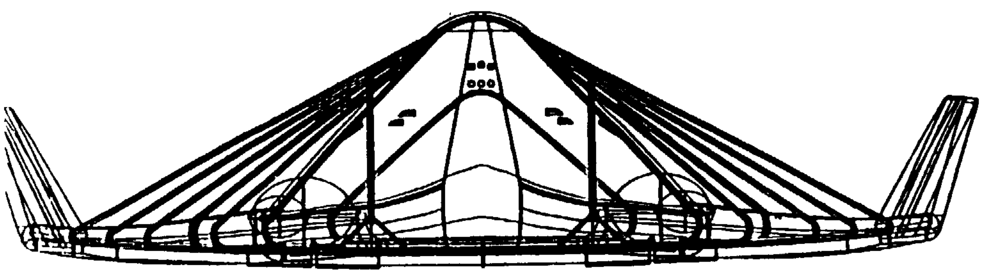
Ref	x, meters	Max press @ lip, Pa	Avg Skin pressure, Pa
0	3.72	3758	N/A
90	0.00	2827	1910
99	-3.45	2310	1841
108	-6.91	2158	1779
117	-10.36	2041	1717
126	-13.82	1931	1662
135	-17.27	1862	1600
144	-20.72	1786	1544
153	-24.18	1717	1489
162	-27.63	1655	1441
171	-31.08	0.00	0.00

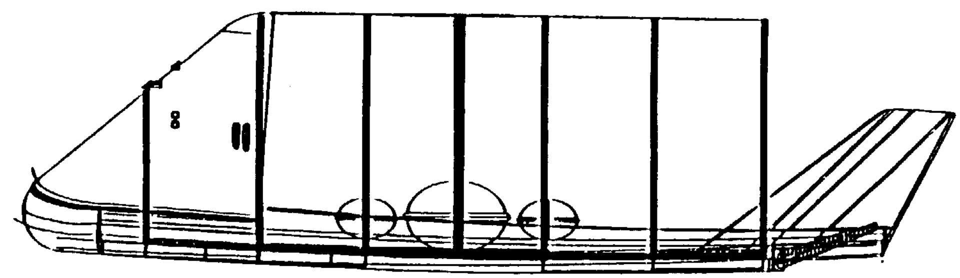
Preliminary Structural Design Assumptions

High L/D Aerobrake Structural Configuration

The chart shown below presents the structural configuration of the frames for the aerobrake. Note that the spacing of the frames was not optimized in the structural analysis. The configuration was fixed as shown for the analysis.

High L/D Aerobrake Structural Configuration





Aerobrake contract/configuration/bew/21Sep90

High L/D Aerobrake Structures

The figures below present the results of the structural sizing exercise. Due to the strength of the Ti3Al "egg crate" core and the SiC/Ti3Al face sheets, the dimensions for the shell panels were driven by minimum gage requirements (assumed as 0.8 mm) rather than maximum stress. This suggests that a lower strength material with lower density (perhaps with thermal barrier coating) may prove to be optimal. Such a trade was not analyzed.

Structure for attachment of the payload to the aerobrake structure was not considered in the mass estimates.

The core thickness was determined to be 25 mm, sandwiched between 0.8 mm (minimum gage) face sheets. Core densiy was taken as 160 kg/m3 (10 pcf).

All dimensions in the facing page figures are in millimeters.

ADVANCED CIVIL SPACE SYSTEMS

High L/D Aerobrake Structures

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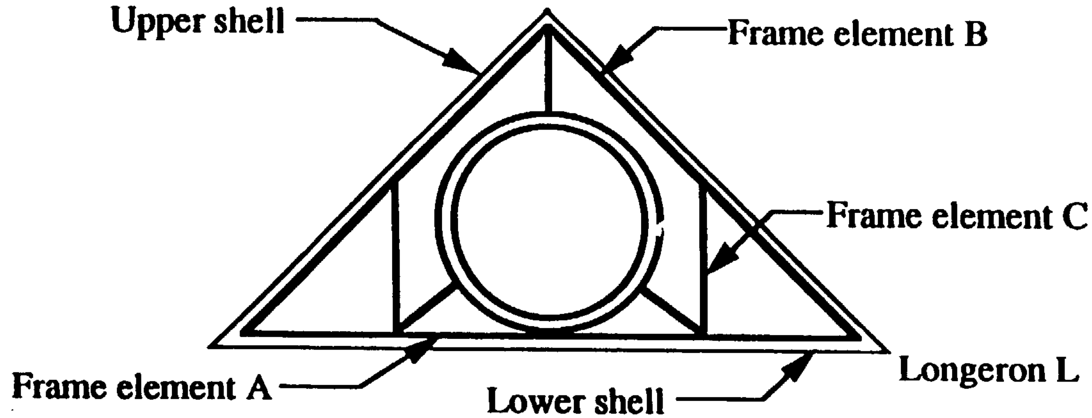
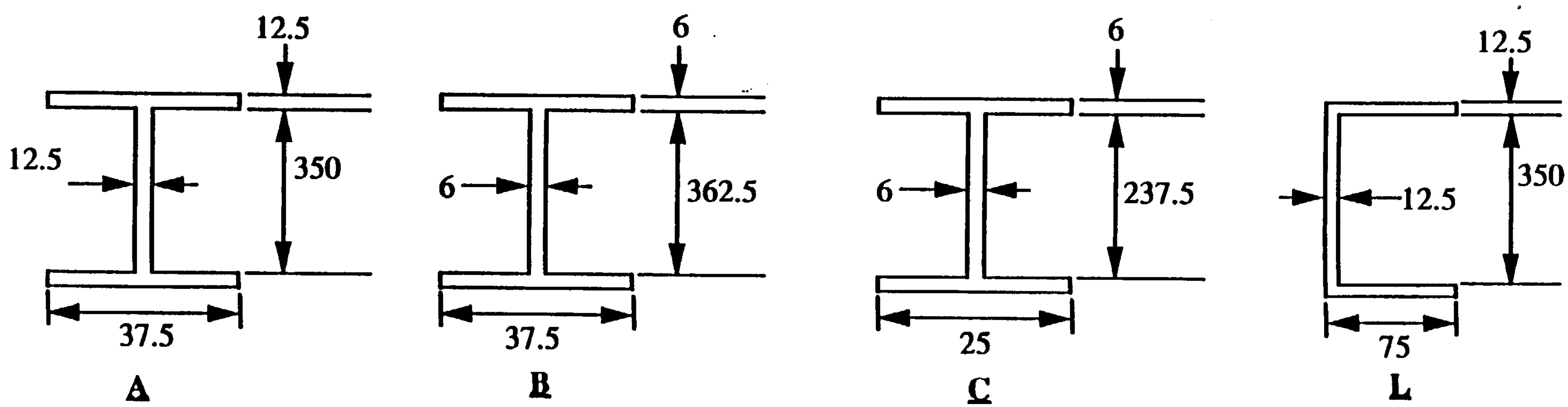
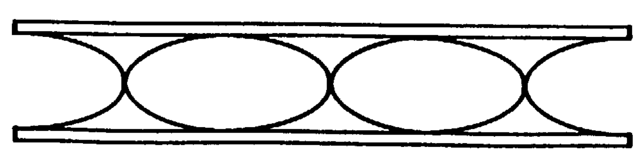


Sizing Results





0.8 mm face sheets

Lower shell panels

Aerobrake contract/size results/bew/21Sep90

High L/D Aerobrake Structures (Results)

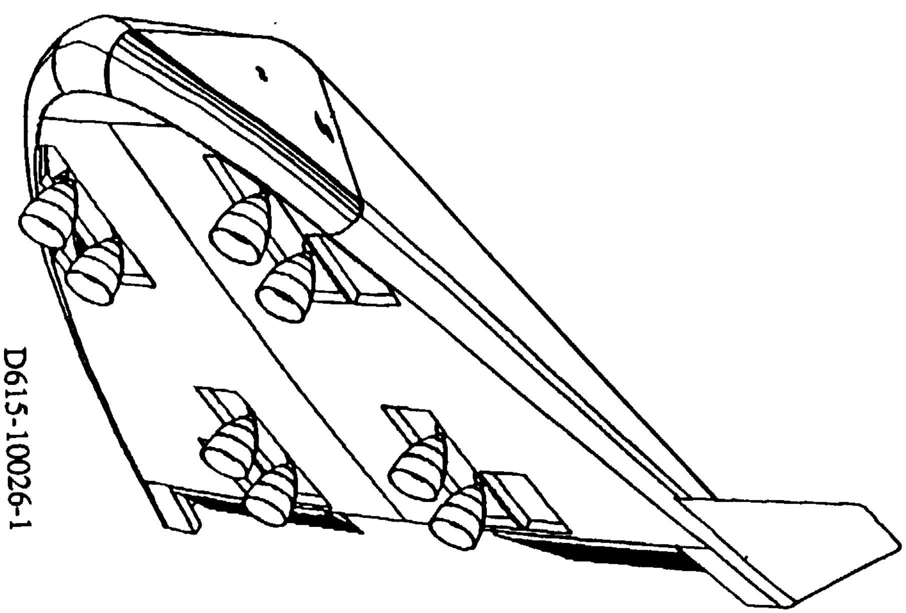
The chart of the facing page presents the results of the structural analysis and mass estimates. The total aerobrake structural weight including thermal barriers is estimated at 16.3 metric tonnes. Of that total, approximately 14.8 tonnes is structure.

The thermal barrier, at 1.4 tonnes, includes coatings and insulation with a strain isolation pad between the coating and the structural element.

This estimate is approximately 15% higher than the initial MEV mass estimate.

The total mass shown corresponds to the cargo expendable mode for the high l/d descent aerobrake.

High L/D Aerobrake Structures



Results

Structure	Mass (kg)
Frame / Truss	3,675
Lower surface shell	6,686
Upper surface shell	2,768
Winglets	1,214
Flaps	440
Total structure weight	14,783
Insulation	416
Coatings	1,106
Total aerobrake weight	16,305

Aerobrake contract/results/bew/21Sep90

High L/D Aerobrake Structures Mass Estimates

The table on the facing page compares the mass estimate of the structural analysis with other methods of estimating the aerobrake structural mass.

The approximate group weights method represents an estimate based on areas of the fuselage (upper shell) and the lifting surface (lower shell). This method makes use of statistical data for aircraft consolidated into one multiplying factor for different types of aircraft. It is intended to give a first approximation of mass for conceptual aircraft without requiring exhaustive calculations. As such, it is not as accurate or precise as most weight estimation methods, especially for extraterrestrial vehicles.

The configuration chosen from preliminary Assured Crew Rescue Vehicle (ACRV) studies represents a vehicle which performs maneuvers similar to those expected for the aerobrake. It also has a comparable lift-to-weight ratio. The configuration was scaled to have dimensions similar to those of the aerobrake.

The initial preliminary weight estimate for the MEV high L/D aerobrake is also included.

Note that the range of the structural analysis (with a 10%) uncertainty parallels the values corresponding to the other methods.

High L/D Aerobrake Structures Mass Estimates

Comparison of mass estimates by method

Estimate / method	Aerobrake structure mass, kg
Approximate group weights (aircraft)	17,646
Scaled ACRV concept (configuration CV-54)	13,134
Initial MEV mass statement	13,774
Structural analysis	16,305
Structural analysis - 10% uncertainty	14,675
Structural analysis +10% uncertainty	17,935

High L/D Structures Key Findings

The chart of the facing page presents the key findings of the structural analysis of the high l/d aerobrake analysis.

High L/D Structures Key Findings

- Aerobrake structure mass is estimated at 16,305 kg.

- For the case analyzed (a=2g) loads are relatively low; assumed structural configuration is adequate. However, for more severe loading cases, structure efficiency can be increased:

    - Shortening load paths will provide lighter elements
    - Long load paths more efficient in tension
    - Optimize element spacing for weight efficiency
    - Frames w/skin & stringer configuration may compare favorably

- For a minimum mass design, largest deflections are near the line of symmetry rather than at the edges.

- Due to the high strength of the Ti3Al "egg crate", face sheet sizes are driven by minimum gage rather than maximum stress. Light weight materials with lower strength may compare favorably.

- Materials for the vehicle represent a major technology development need.

High L/D Aerobrake Structures

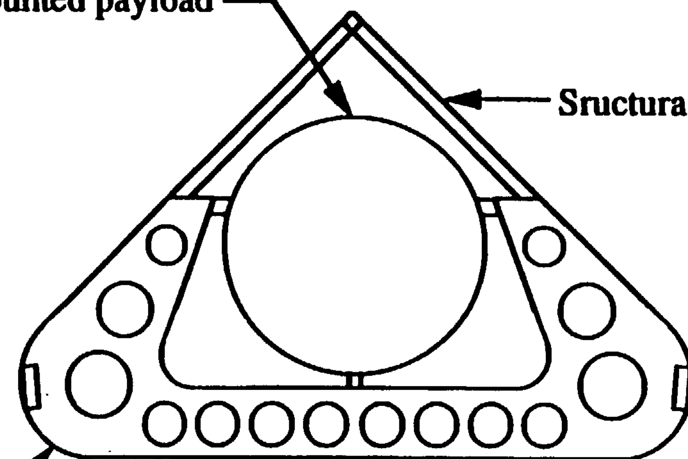
The figure on the chart below presents a candidate structural system for the high l/d aerobrake.

While the structural configuration analyzed was adequate for the loading case of descent, it was not analyzed for other loading conditions. For instance, if the aerobrake is launched fully assembled on an external tank (as has been proposed), design loading conditions may change. The structural system shown below is based on thoroughly analyzed concepts such as Shuttle C.

A cursory inspection of the original concept indicates a need to shorten load paths and ensure that tension dominates the two-force members (for minimum mass). The proposed concept below is intended to reduce these initial concerns.

High L/D Aerobrake Structures

Proposed Structure: Typical cross-section



Proven trunnion mounted structural system

Similar to Shuttle mounted concepts such as Shuttle-C

Frame web holes increase mass savings

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Aerobrake Design Concerns

- Many state-of-the-art advances are needed to support on-orbit assembly and checkout
  - Robotics
  - Non-destructive inspection
  - "Smart" structure
  - Joint closure

- Advanced Thermal Protection System (TPS) materials are essential for C3 > 30; temperatures > 1800°C will probably require a mass penalty

G-GRM/2H895/AEROBRAKE STUDY/DISK 4/O/165-0/11:00A

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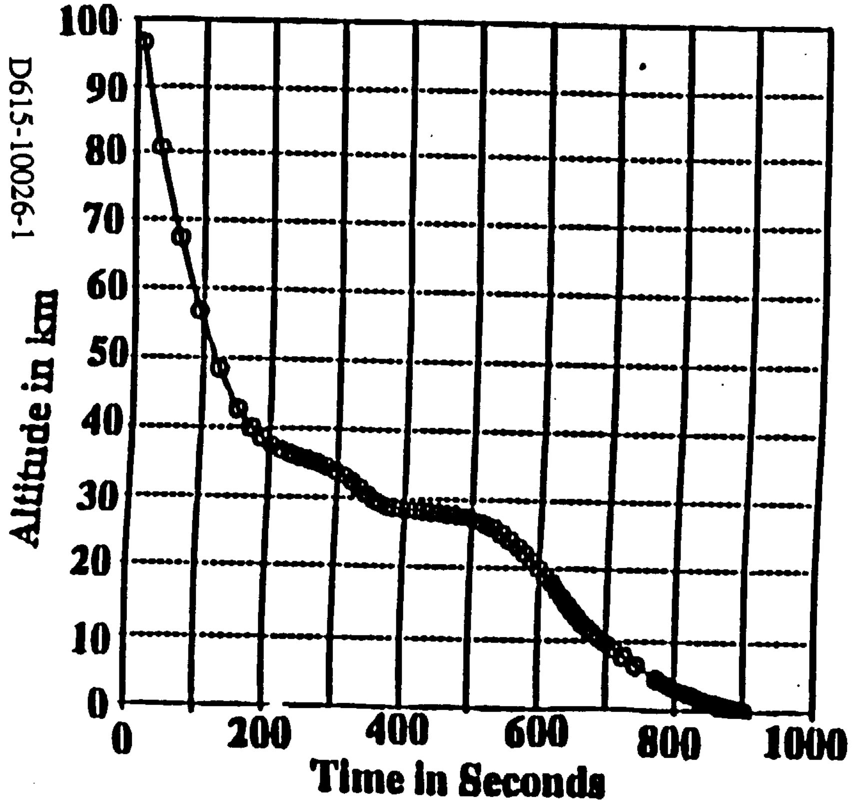
The aerobrake configuration utilized in this analysis is the same as previous. With an L/D of 1.0 the flight time is 2 times greater than for an L/D = 0.5, thereby resulting in range increase of 50%.

Landing Analysis

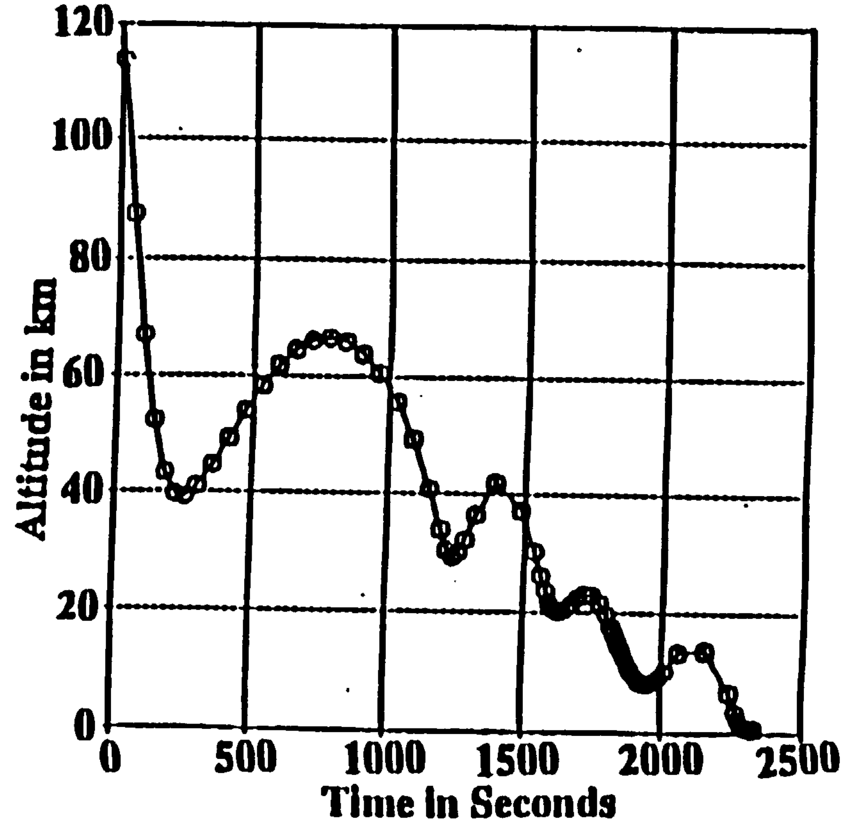
Range Effect of L/D

Entry Mass	81 mt
Isp	470 sec
Ref Area	471 m^2

L/D = 1/2



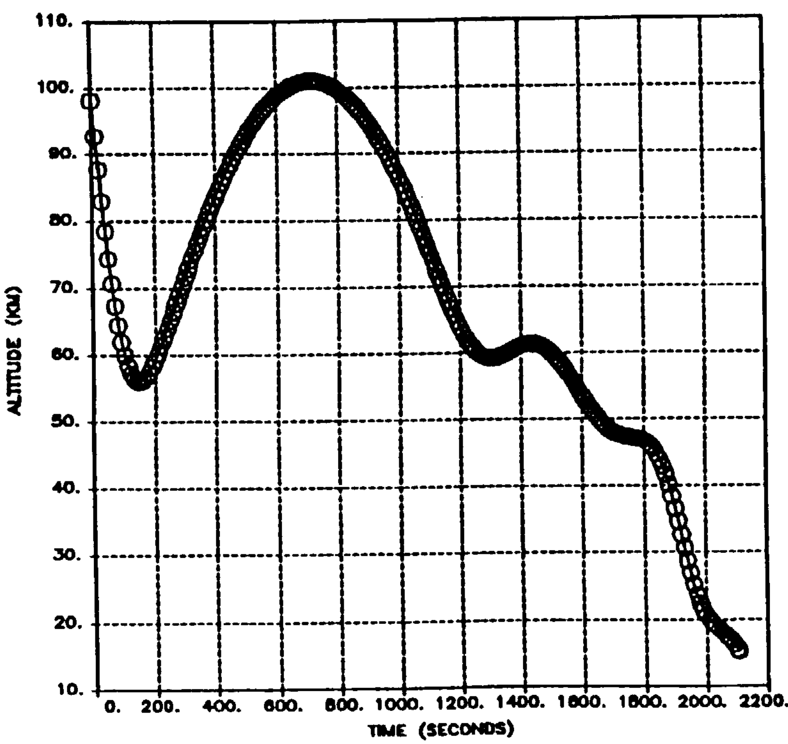
L/D = 1.0

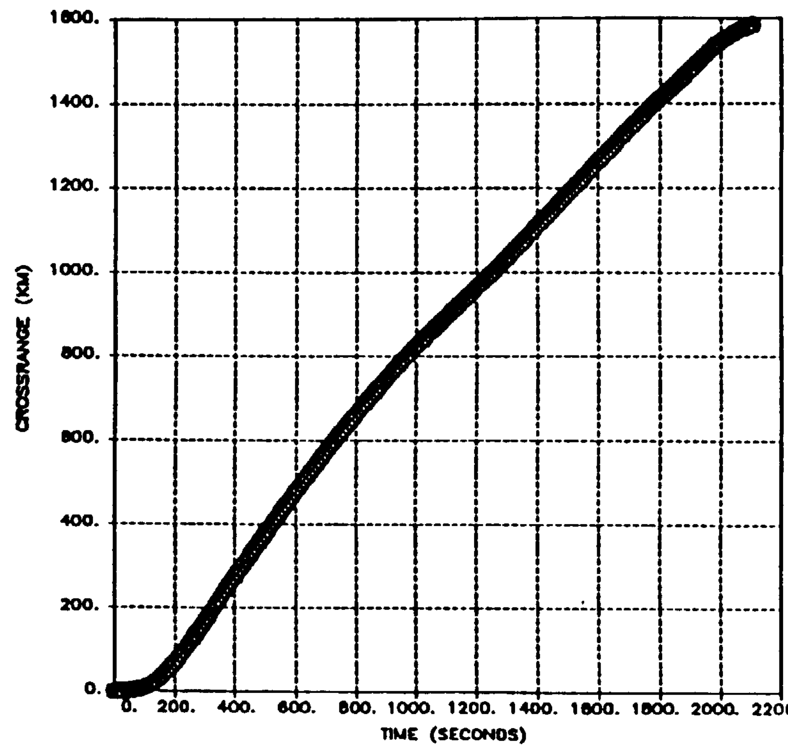


GRM/2H895/AEROBRAKE STUDY/DISK 3/M/166-0/10:00A

The following plots were generated using the OTIS (Optimal Trajectories by Implicit Simulation) program. The plots show the flight path of a vehicle that will give the maximum crossrange and they show the crossrange. The plots were done for a vehicle flying at a lift to drag ration of one and a lift to drag ration of one-half.

Lift to Drag Ratio = 1.0





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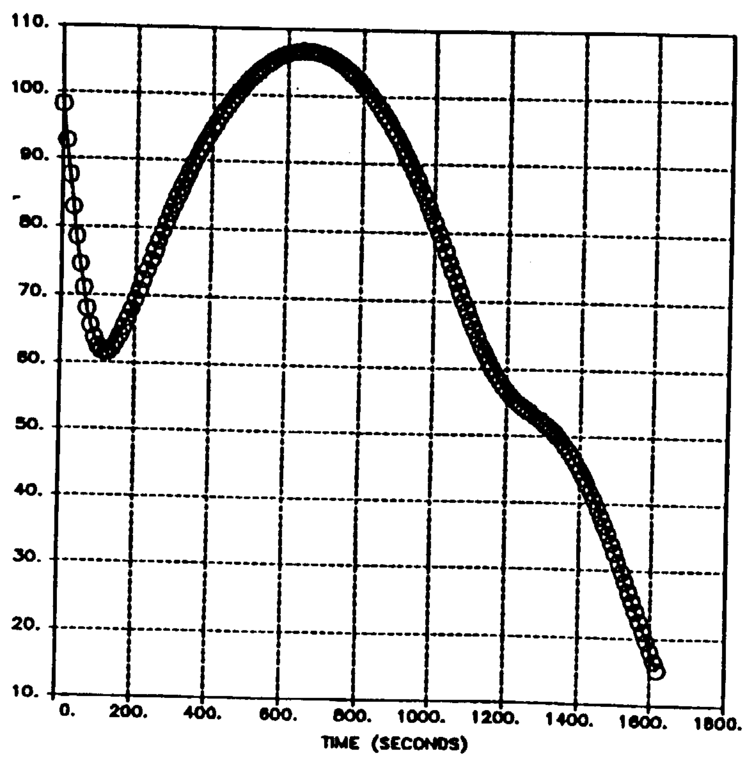
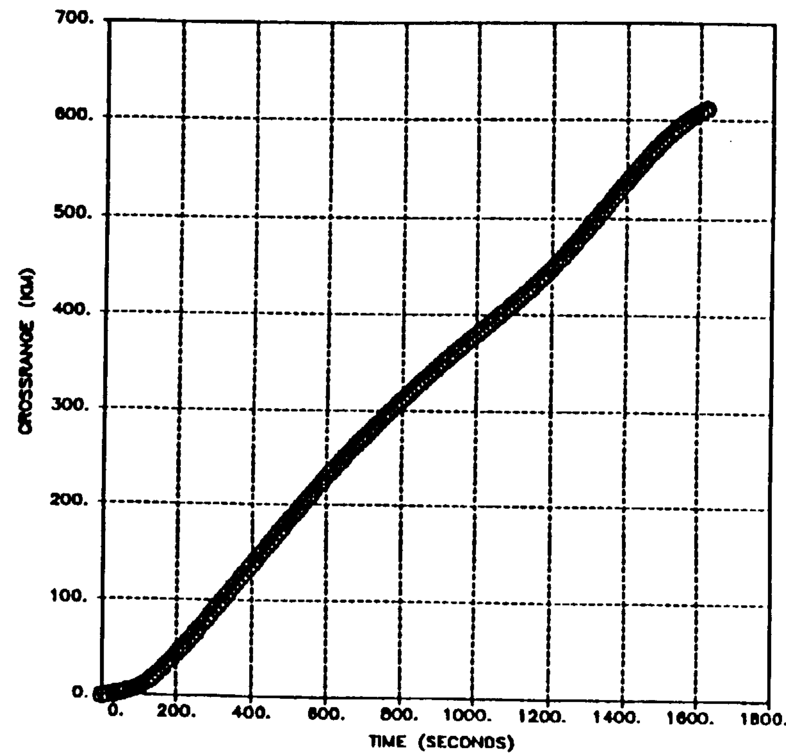
Lift to Drag Ratio = 0.5





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For a range of L/D from 0.65 to 1.0, the ideal delta velocity range is from 1400 m/s to 1200 m/s for landing at a 5 km altitude. In the case of an aeroflare with two different reference areas of 470 and 750 m^2 with a cutoff velocity 0.02 the ideal delta velocity is approximately 900m per second. This results in a delta velocity reduction due to using the aeroflare from 200-300 m/s for L/D = 1.0.

Landing Analysis - Aeroflare
Ideal Delta Velocity

Reference Area = 471 sq. m

Thrust = 420 kn

ESA = 11.4 km

Aerobrake Drop = 6.79 km

Cutoff Velocity = .1 km/s

Ideal Del Velocity = 529 m/s

Reference = 750 sq. m

Thrust = 420 kn

ESA = 11 km

Aerobrake Drop = 8.52

Cutoff Velocity = 0.02

Ideal Del Velocity = 918.5

G GRM/2H895/AEROBRAKE STUDY/DISK 6/H/165-0/10:00A

Mars Descent - Pinpoint Landing

This data was generated to evaluate the requirements to make a land at a specific position. The ground coordinates were chosen as +7.0 degrees latitude and 80 degrees longitude. The initial orbit conditions were taken from the 2016 Boeing Nominal mission PLANET run. The initial latitude and longitude coordinates were (-37.77, 64.08) at start of descent; the initial apoapsis altitude, periapsis altitude and inclination were 37000 km, 35 km, and 50 degrees respectively. The trajectory was constrained to end at a specific location. The final latitude was to be between 5 degrees and 10 degrees. The final longitude was to be between 75 degrees and 85 degrees. The vehicle was flown at an angle of attack of 44.35 degrees (this corresponds to an L/D of 1.0). The bank angle was allowed to vary in such a way that the final velocity was minimized. For the data run presented the final velocity was approximately 400 m/sec.

ADVANCED CIVIL SPACE SYSTEMS

Mars Descent - Pinpoint Landing

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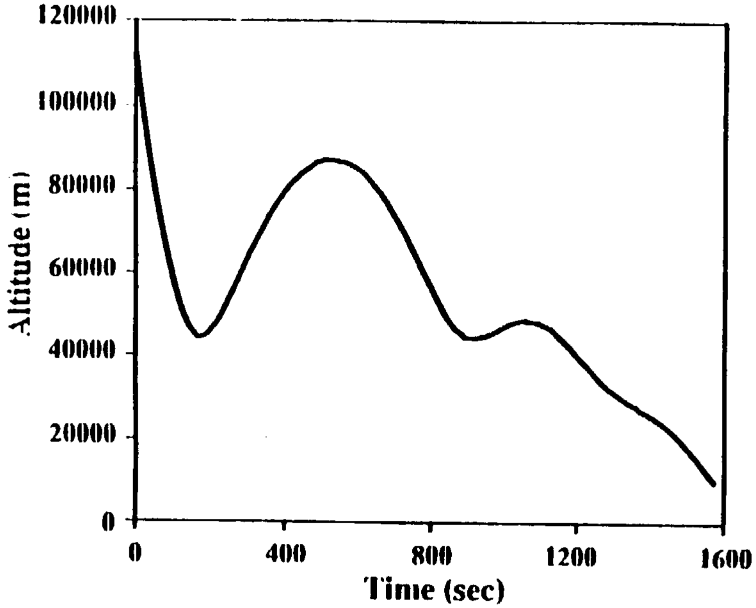
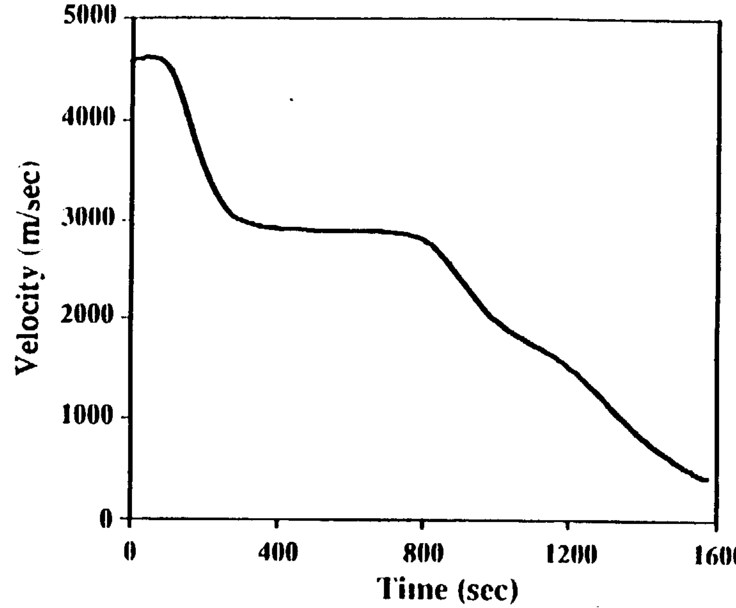
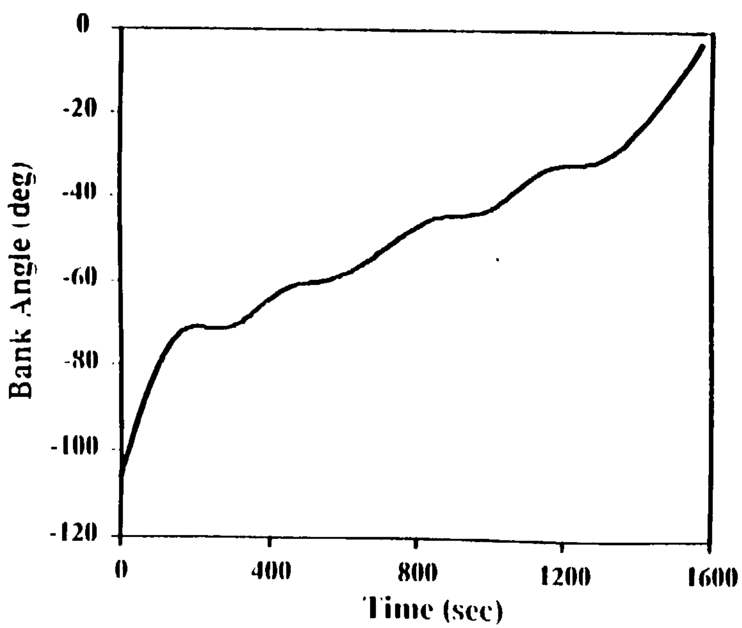
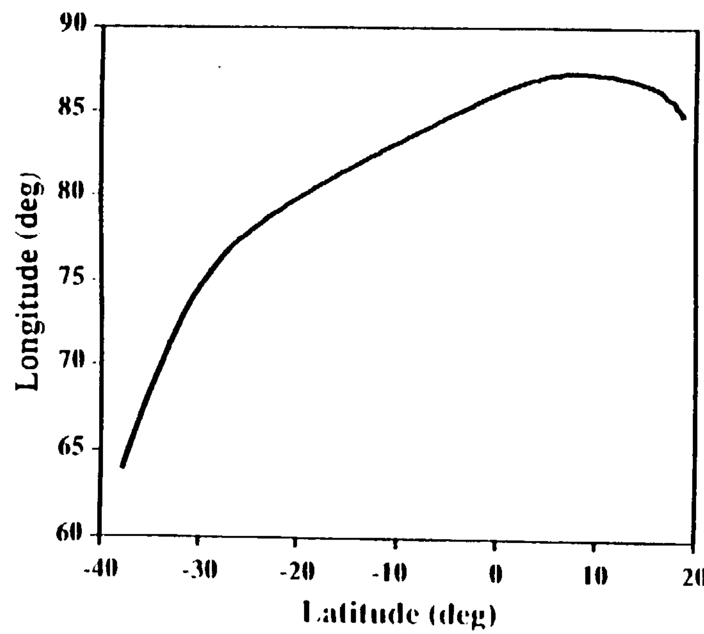








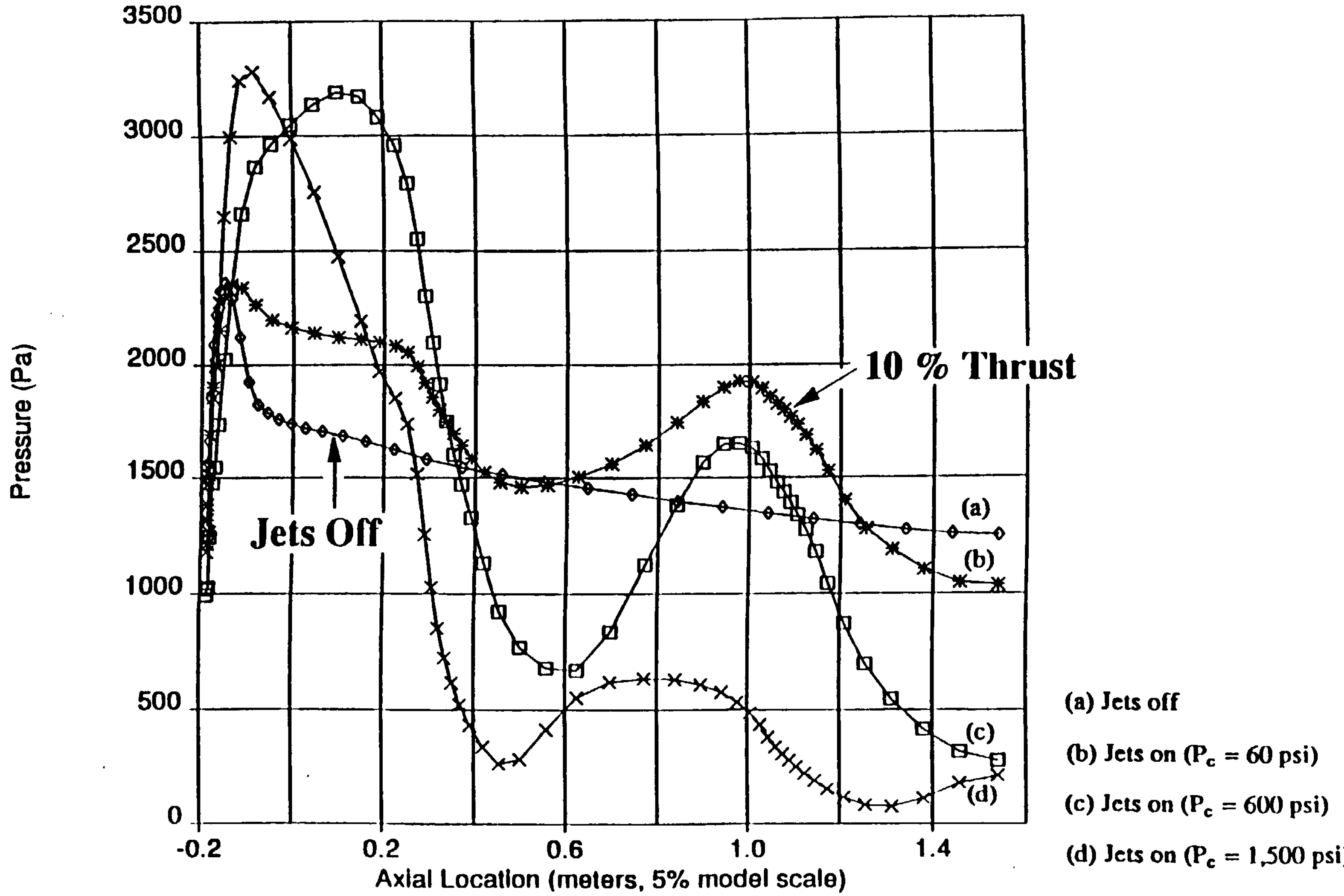
High L/D Descent, 10% Thrust
CFD Solution

Shown on the graph below are preliminary results for a Computational Fluid Dynamics (CFD) analysis of the high L/D aerobrake, right after engine start during a descent trajectory. This analysis was performed using a 3-D Navier-Stokes program (GIFS, developed by Boeing for Edward's AFB Atsronautics Lab). A grid size of 55 x 31 x 41 was used for the analysis. Freestream conditions were taken from a high L/D descent simulation at the time of engine start. These conditions are:

Altitude = 9 km
Velocity = 600 m/sec
Pressure = 252 Pa
Density = 6.66e-3 kg/m^3
Angle of Attack = 45 degrees

The flow was composed of a mixture of ideal gases, which was a freesteam of CO_2 and a rocket exhaust of H_2O. The following assumptions were made for the engines: area ratio (exit/throat) = 200 , nozzle exit diameter = 2.1 meters, chamber temperature = 3500 k, mixture ratio = 6:1 LOx/LH, exhaust gas = H_2O. The chamber pressure was varied to simulate varying thrust levels. Of paticular intrest was the 60 psi chamber pressure case, which aproximated a 10% thrust level. At a 10% thrust level there is not enough control authority and it was felt that there could have been a severe moment placed on the front of the vehicle resulting in control problems. This was not the case, as seen in the results below, which show no severe pressure distribution at the front of the vehicle as in the case of the higher thrust levels.

High L/D Descent, 10% Thrust CFD Solution



Windward centerline pressures.

Preliminary Mars Landing Sites Between +/- 20° Latitude

Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
Tharsis Montes	5°N	100°	9 km	Ascraeus and Pavonis Mons, rill formations, Tharsis Tholus, unnamed crater
	10°N	82°	3-2 km	Tharsis Tholus, Echus Chasma, Fesenkov Crater, head of Kasei Vallis, Lunae Planum (colored soil)
	-10° S	137°	4 km	Mangala Vallis, Memnonia and Sirenum Fossae, edge of Tharsis Montes shield, Aganippe Fossa, Arsia Mons Colored sands
	-15° S	115°	8-9 km	Arsia Mons, Noctis Labyrinthus, Syria Planum, Claritas Fossae, crater area
Sinai Planum	-18°S	76°	1-8 km	Melas Chasma of Valles Marineras (possible access to Valles Marineras floor); Felis, Melas and Solis Dorsii, crater (unnamed) with rills/flows, Lassell Crater, Coprates Catena

ADVANCED CIVIL SPACE SYSTEMS

Preliminary Mars Landing Sites Between +/- 20° Latitude page 2

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Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
South of Eos Chasma	-19°S	49°	3 km	Eos Chasma (part of Valles Marineras, with possible access to the valley floor, accessible places in the valley floor -1 and -2 km) Lassell and Ritchey Craters, Felis Dorsa, crater field some with flow fields, Holden Crater
Hesperia Planum	-16°S	253°	4 km	Tyrrhena Patera (massive flow field from a single source), crater fields, surface cracks and fissures, Terra Tyrrhena area, small mounts
Elysium-Amazonis	0°	180°	0 km	Pettit Crater, Nicholson Crater, surface cracks, Orcus Patera, Cerberus Rupes, colored soils, old craters, Apollinaris Patera, Gusev Crater and flow field, edge of Elysium flow shield, Medusae Fossae, "new" craters in the Elysium flow shield

Preliminary Mars Landing Sites Between +/- 20° Latitude page 3

Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
Amazonis Planitia	15°N	155°	0-3 km	flow area around Olympus Mons, edge of Gordii Dorsum and Eumenides Dorsum formations, crater area east of Pettit Crater
Chryse Planitia	18° N	45°	0-(-1) km	Chryse depression (-3 km), end of Kasei Vallis, Sharonov Crater,Lunae Planum; Nanedi, Shalbatanu , Simud, and Tiu Valles, end of Ares Vallis, craters, colored sands

Preliminary Mars Landing Sites Between +/- 20° Latitude page 4

Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
North of Ganges Catena	-2°S	68°	1 km	Ophir Chasma (part of Valles Marineras, with possible access to the valley floor), Hebes Chasma, Echus Chasma, Juventae Chasma, Ophir Planum, Lunae Planum, crater field, colored soil
Elysium Planitia	19°N	226°	0 km	Hephaestus Fossae, Elysium Fossae,Elysium Mons, Albor Tholus, Eddie Crater with interior formation, colored sands
	19° N	197.5°	2-3 km	Elysium Mons, Elysium Fossae, Ocrus Patera, Cerberus Rupes, Lockyer Crater, Phlegra Montes, colored sands, craters, old and "new"

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Mars Ascent Conditions - High Density

Initial mass = 113,000 kg

Final mass = 41,800 kg

Reference area = 600 m^2

Isp = 475 sec

Velocity losses due to drag = 217 m/s

Final altitude = 98,800 m

Apoapsis altitude = 100 km

Periapsis altitude = 90 km

Inclination = 2.32°

Argument of periapsis = 229°

Right ascension = -0.64°

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ADVANCED CIVIL SPACE SYSTEMS

Reusable MEV; Ascent - High Density

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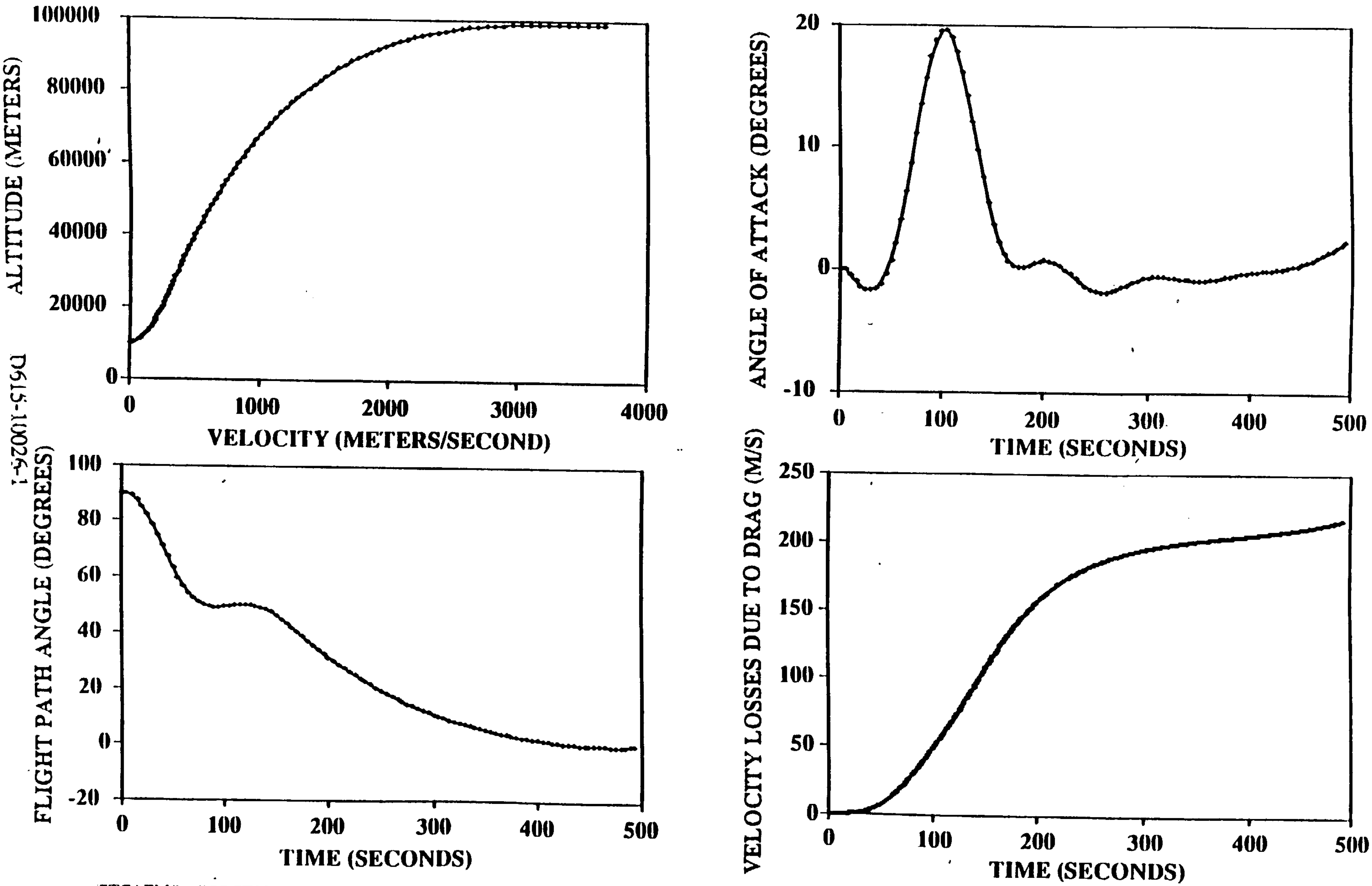

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Mars Ascent Conditions - Average Density

Initial mass = 113,000 kg

Final mass = 42,500 kg

Reference area = 600 m^2

Isp = 475 /sec

Velocity losses due to drag = 178 m/s

Final altitude = 98,300 m

Apoapsis altitude = 100 km

Periapsis altitude = 92 km

Inclination = 2.31°

Argument of periapsis = 245°

Right ascension = -0.65°

ADVANCED CIVIL SPACE SYSTEMS

Reusable MEV; Ascent - Average Density

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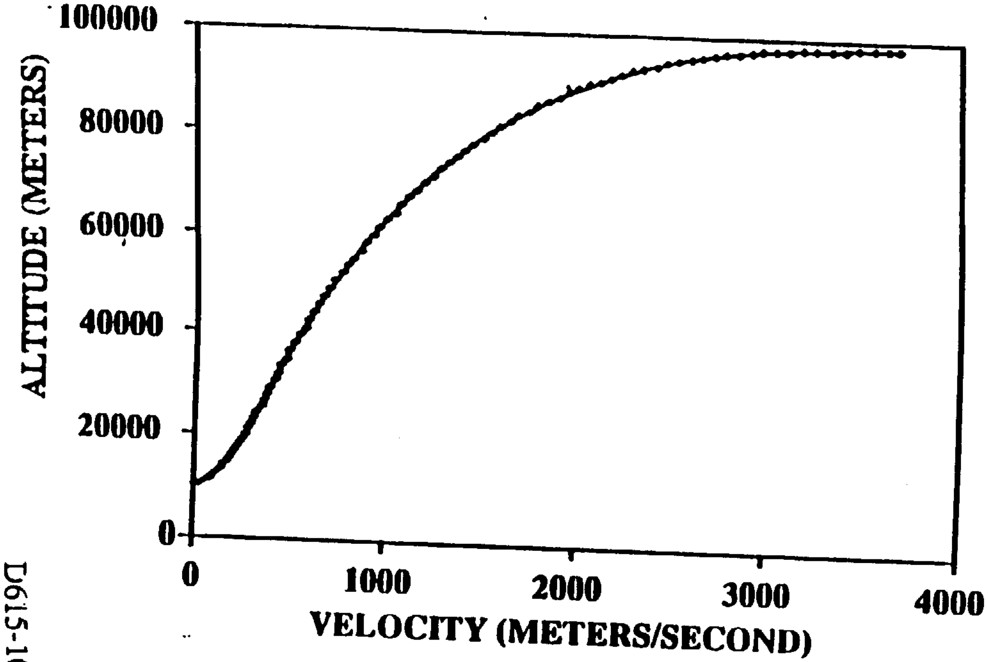

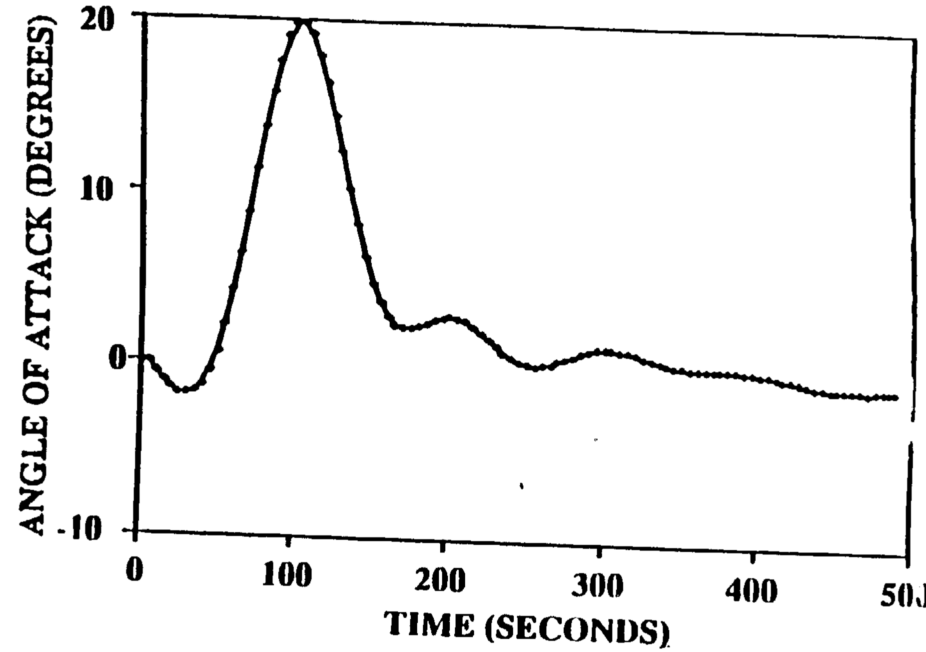

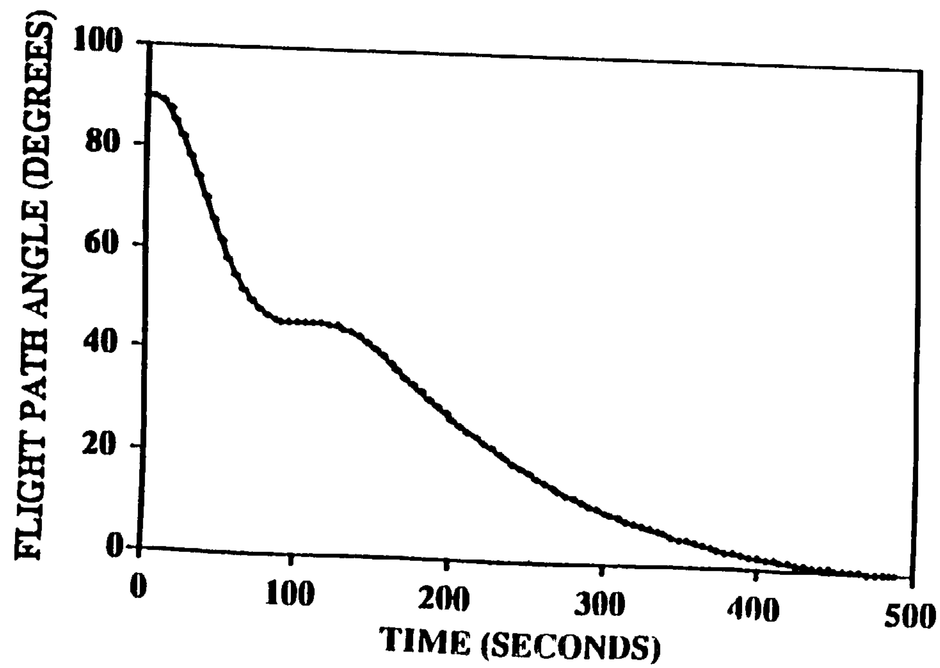

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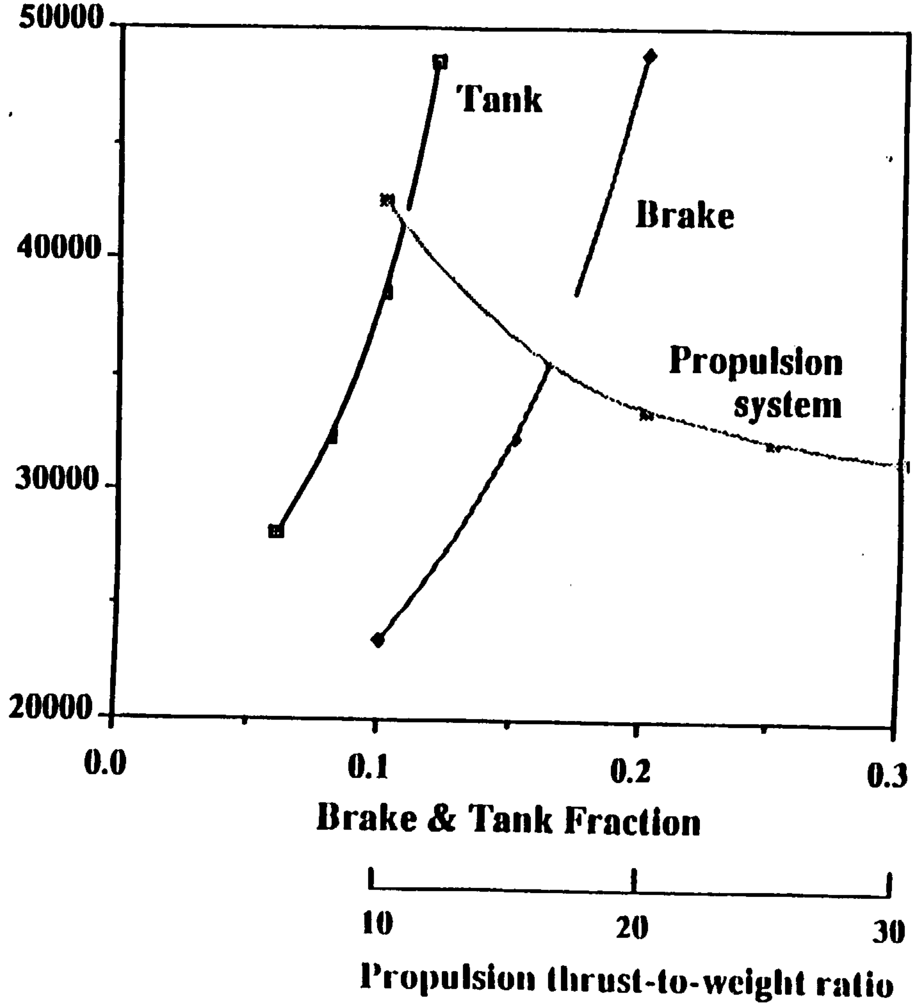
Reusable MEV Status

Reusable MEV Parametrics



Assumptions

- Landing propellant and ascent H2 supplied in Mars orbit
- Ascent O2 from mars surface.
- 470 Isp
- Down payload 29.5 t. (25 t. + 4.5 t. c/m)
- Ascent payload 4.5 t. crew module
- Descent: aerobrake + 1200 m/sec
- Ascent ΔV 5500 m/sec
- Mixture ratio 6:1

Configuration Issues

- Location of payload at c.g. for landing
- Payload removal method
- Location of crew module for landing visibility
- Location of engines for landing and ascent
- Location and arrangement of tanks
- Landing gear design and placement for landing and ascent

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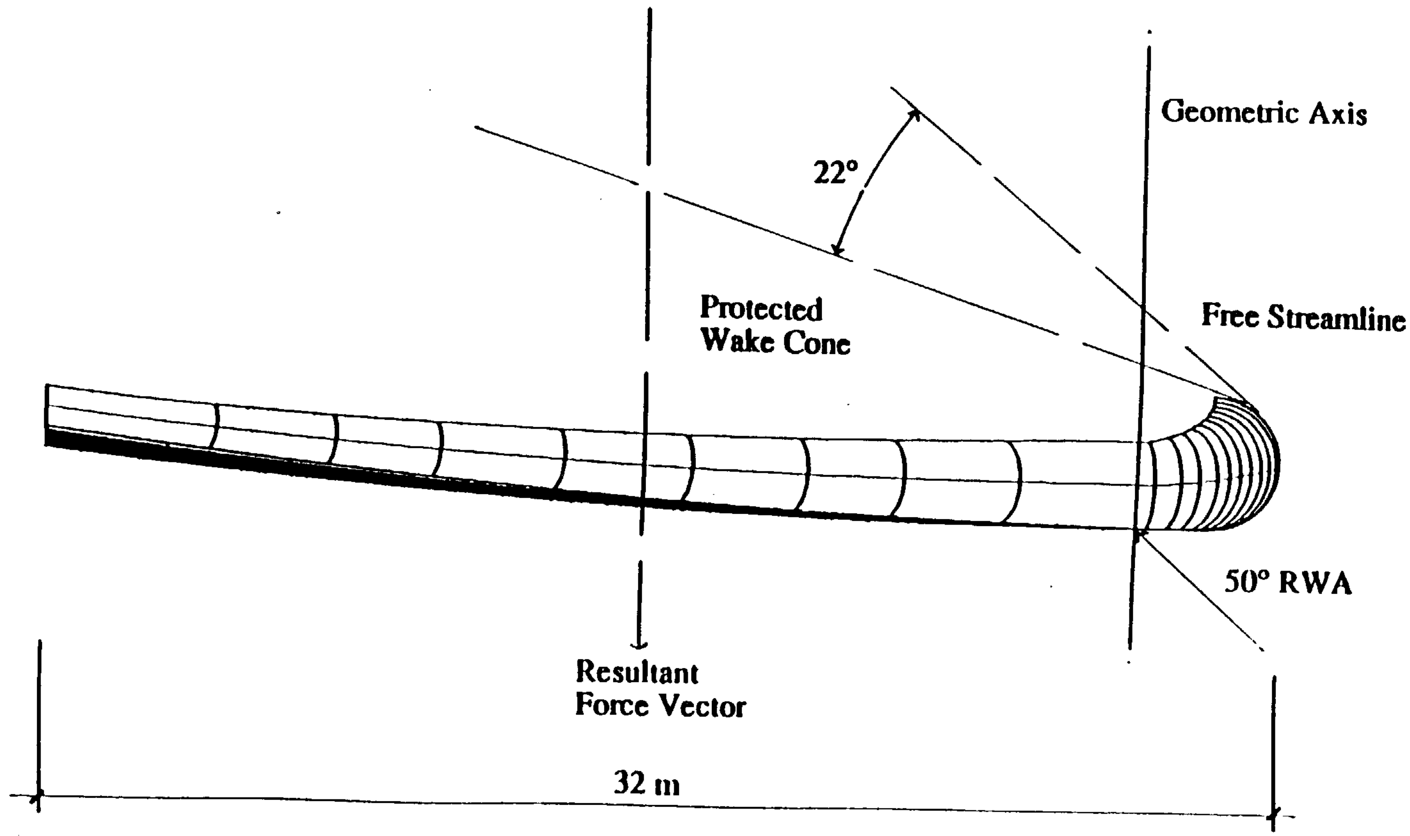
High L/D Aerobraking Constraints

Shown on the facing page are constraints as applied to the high L/D aerobrake, which were used to configure the reusable MEV. Aerobraking constraints include resultant force vector and protected wake cone, which impact the location of the MEV within the aerobrake.

High L/D (1.1) Aerobraking Constraints



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Flowfield Analysis Assumptions

Flowfield assumptions:

- Altitude = 9 km
- Velocity = 600 m/sec
- Pressure = 252 Pa
- Density = 6.66 -3 kg/m^3
- Angle of Attack = 45 degrees
- Mixture of ideal gases in plume
  - freesteam CO_2
  - rocket exhaust H_2O

Rocket engine assumptions:

- Area Ratio (exit/throat) = 200
- Nozzle exit diameter = 2.1 meters
- Chamber Pressure = 10.3 MPa
- Chamber Temperature = 3500 k
- Mixture Ratio LOx/LH = 6:1
- Exhaust = H_2O

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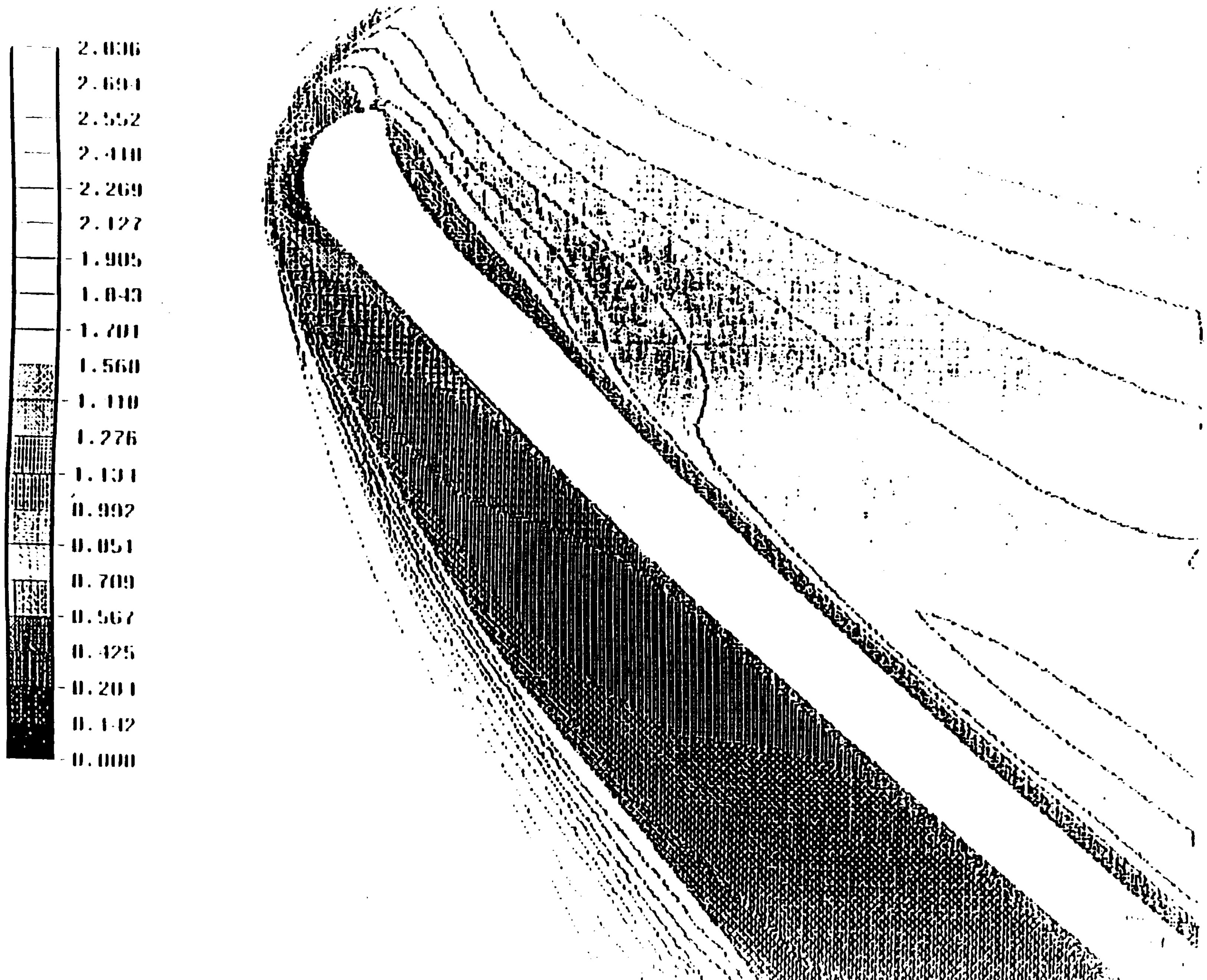
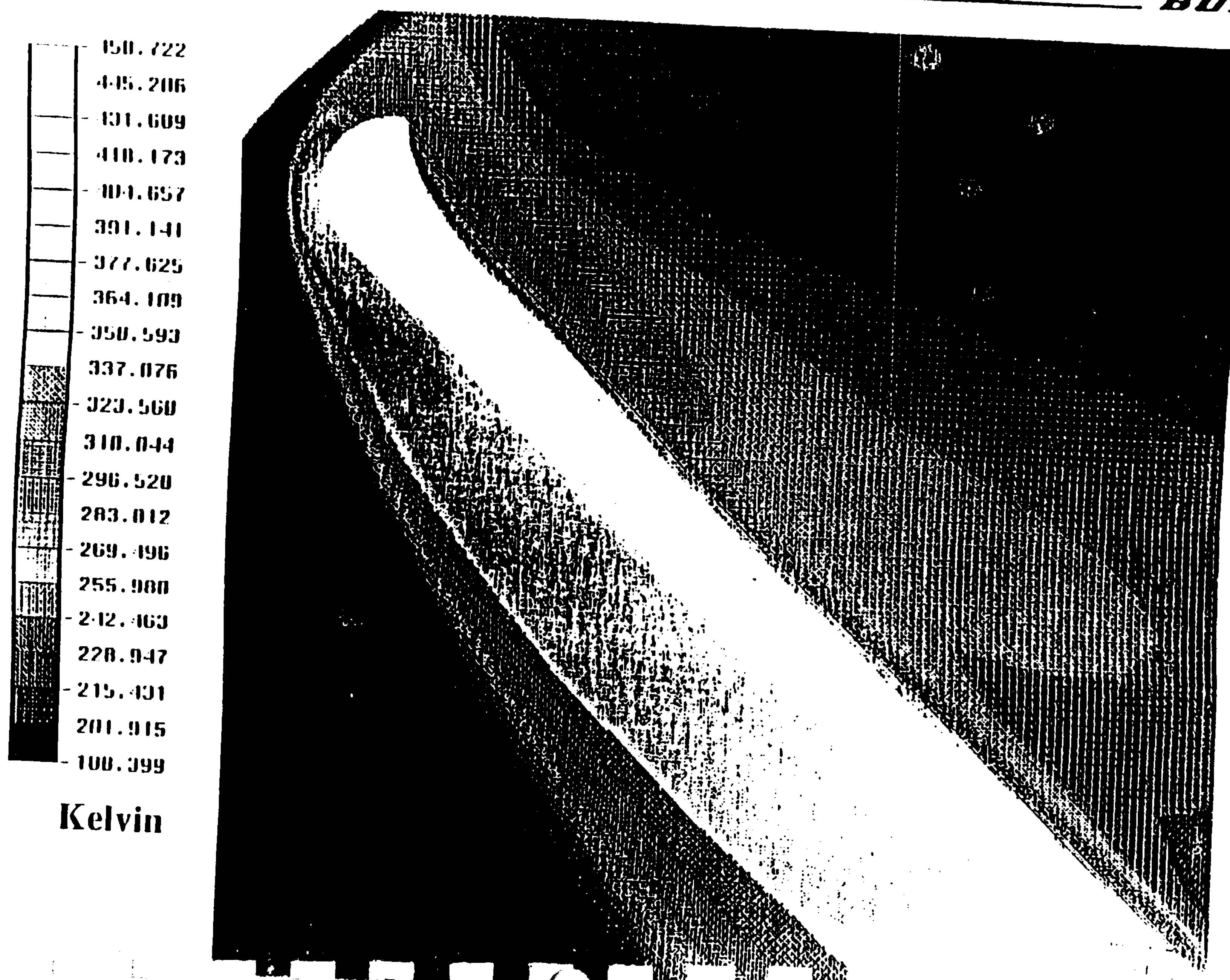
Mach Number Contours - Symmetry Plane

ADVANCED CIVIL SPACE SYSTEMS

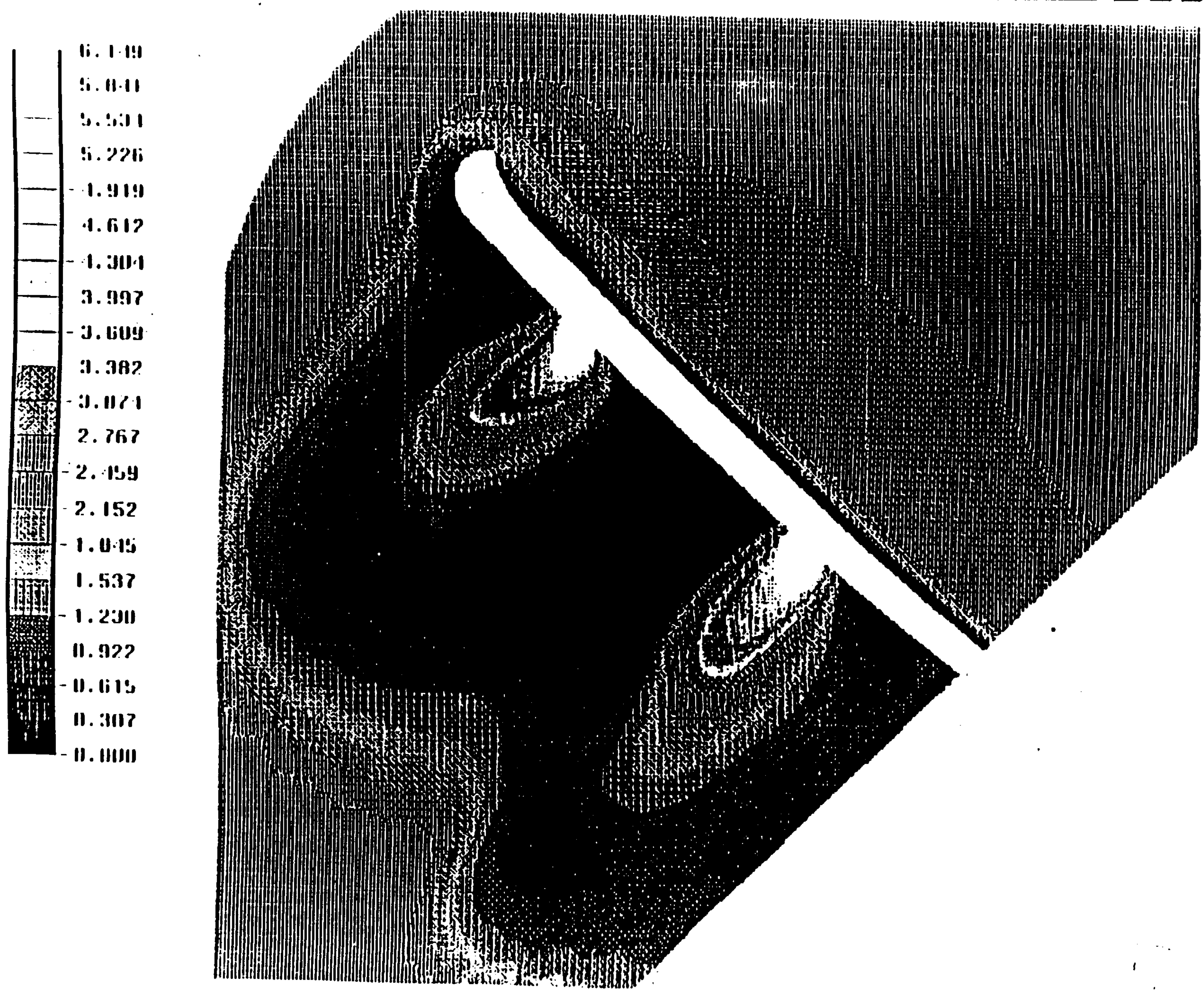
BOEING



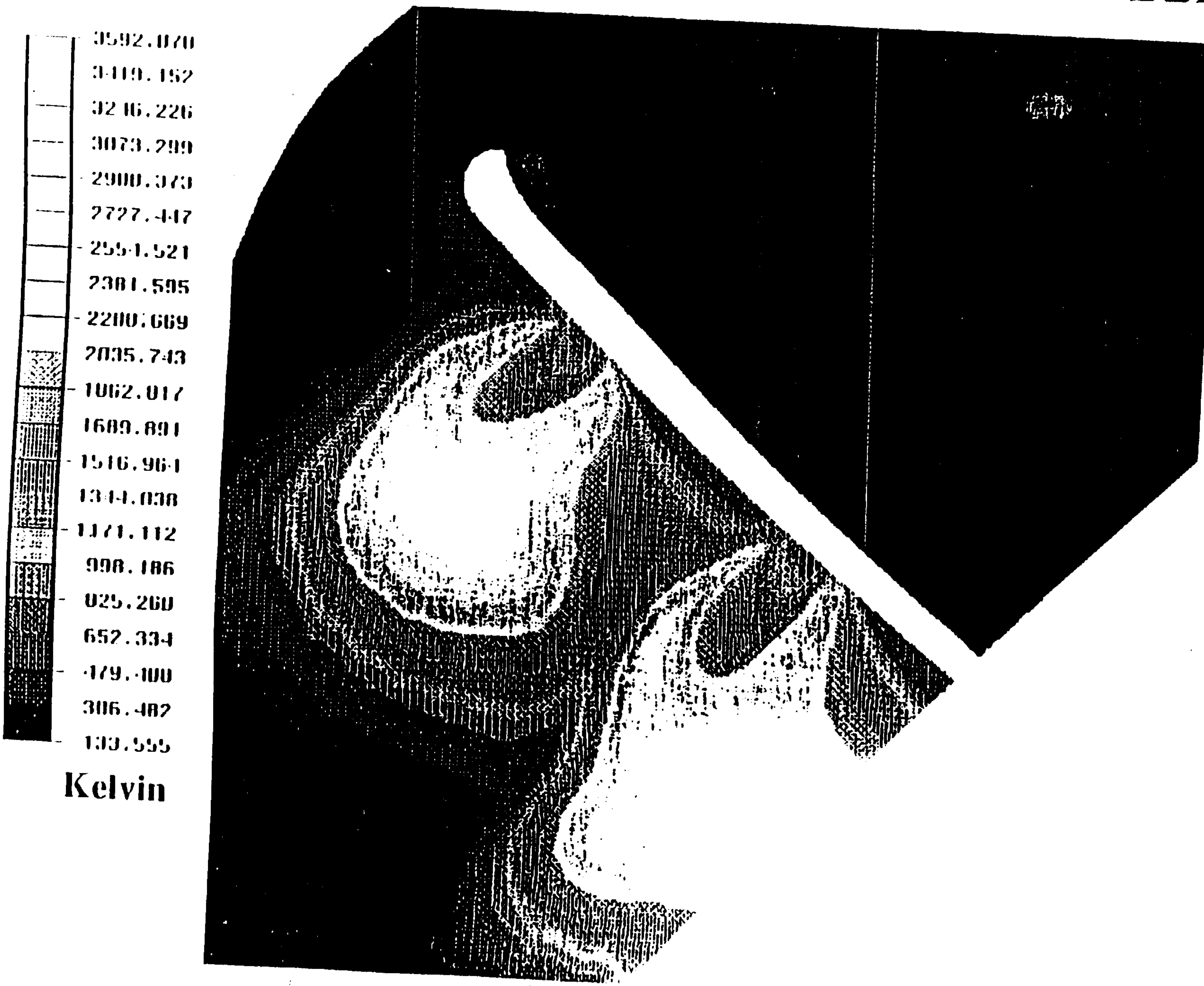
Temperature Contours - Symmetry Plane



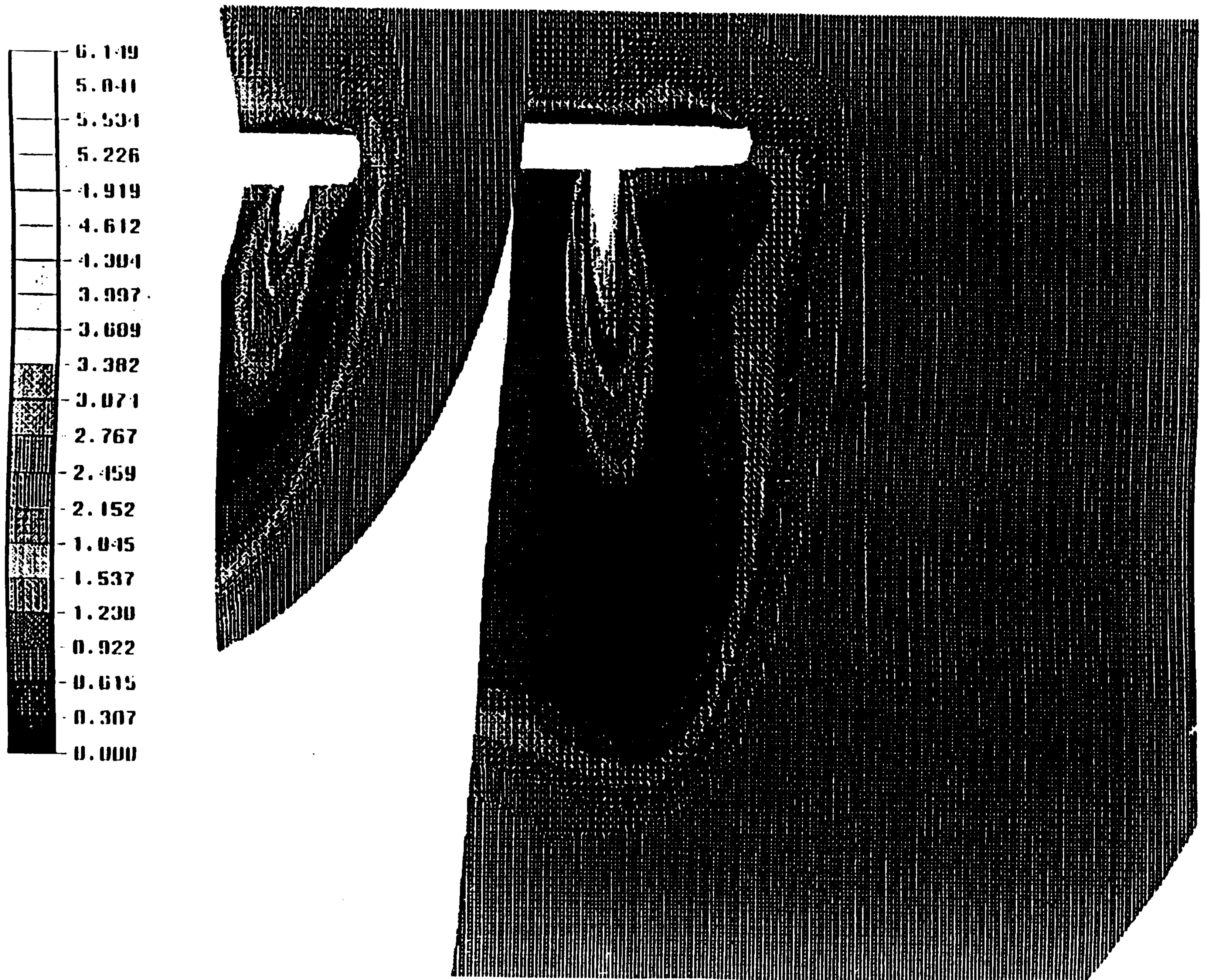
Mach Number Contours - Longitudinal Jet Plane



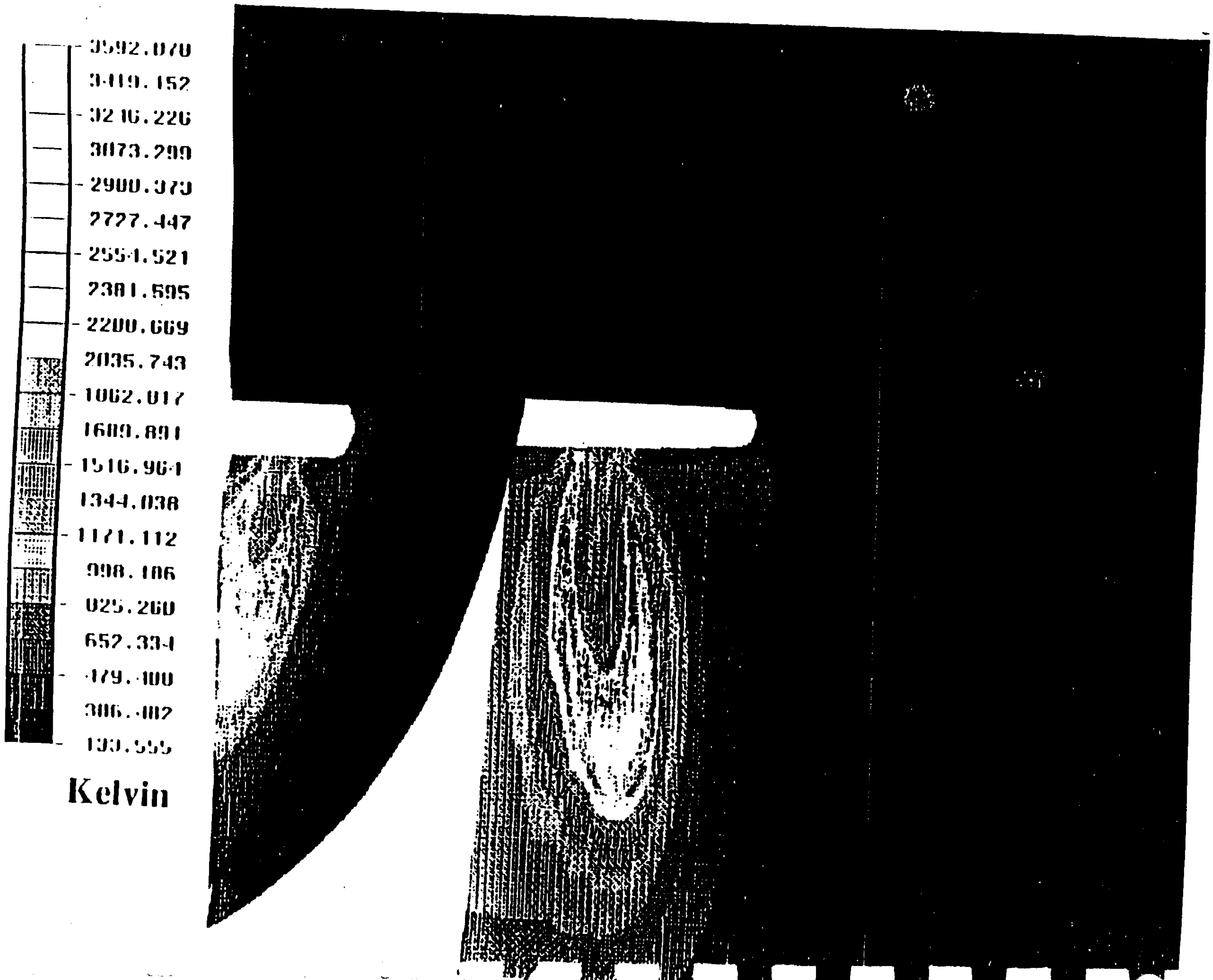
Temperature Contours - Longitudinal Jet Plane



Mach Number Contours - Transverse Jet Planes



Temperature Contours - Transverse Jet Planes



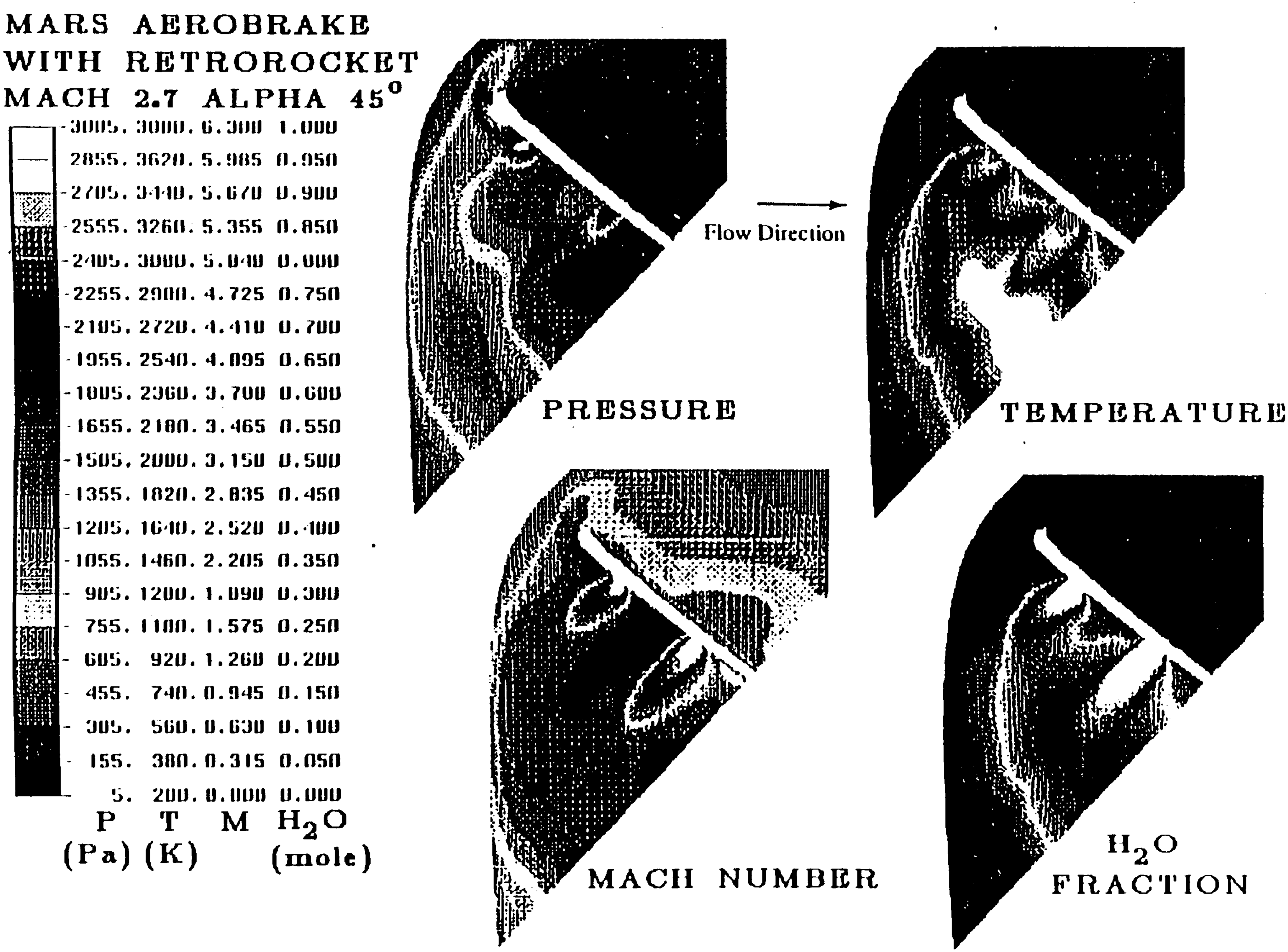


Figure 4. Flowfield solution with jets (symmetry plane/jet centerline).

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Mars Aerobrake Pressure Distribution

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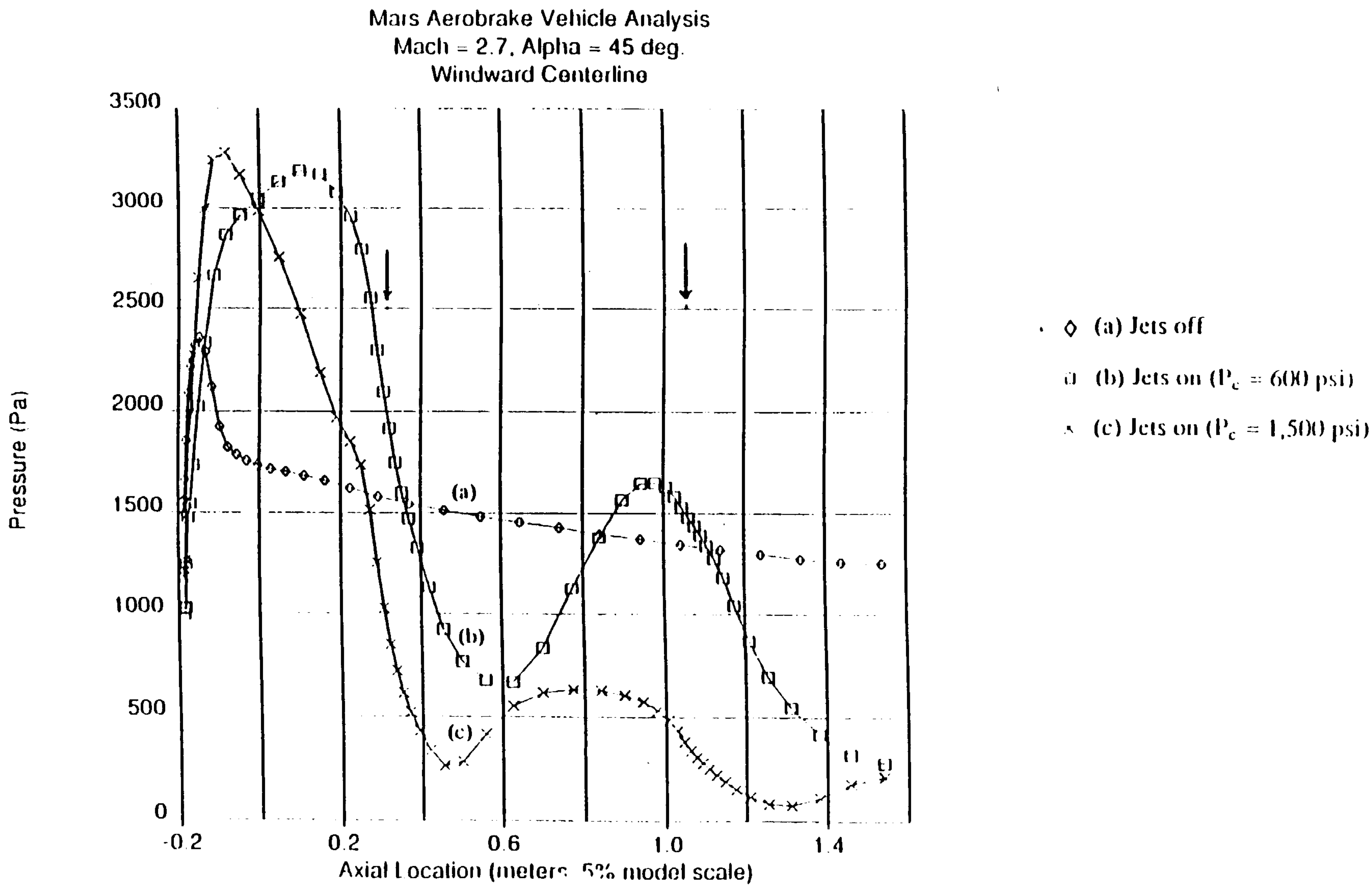

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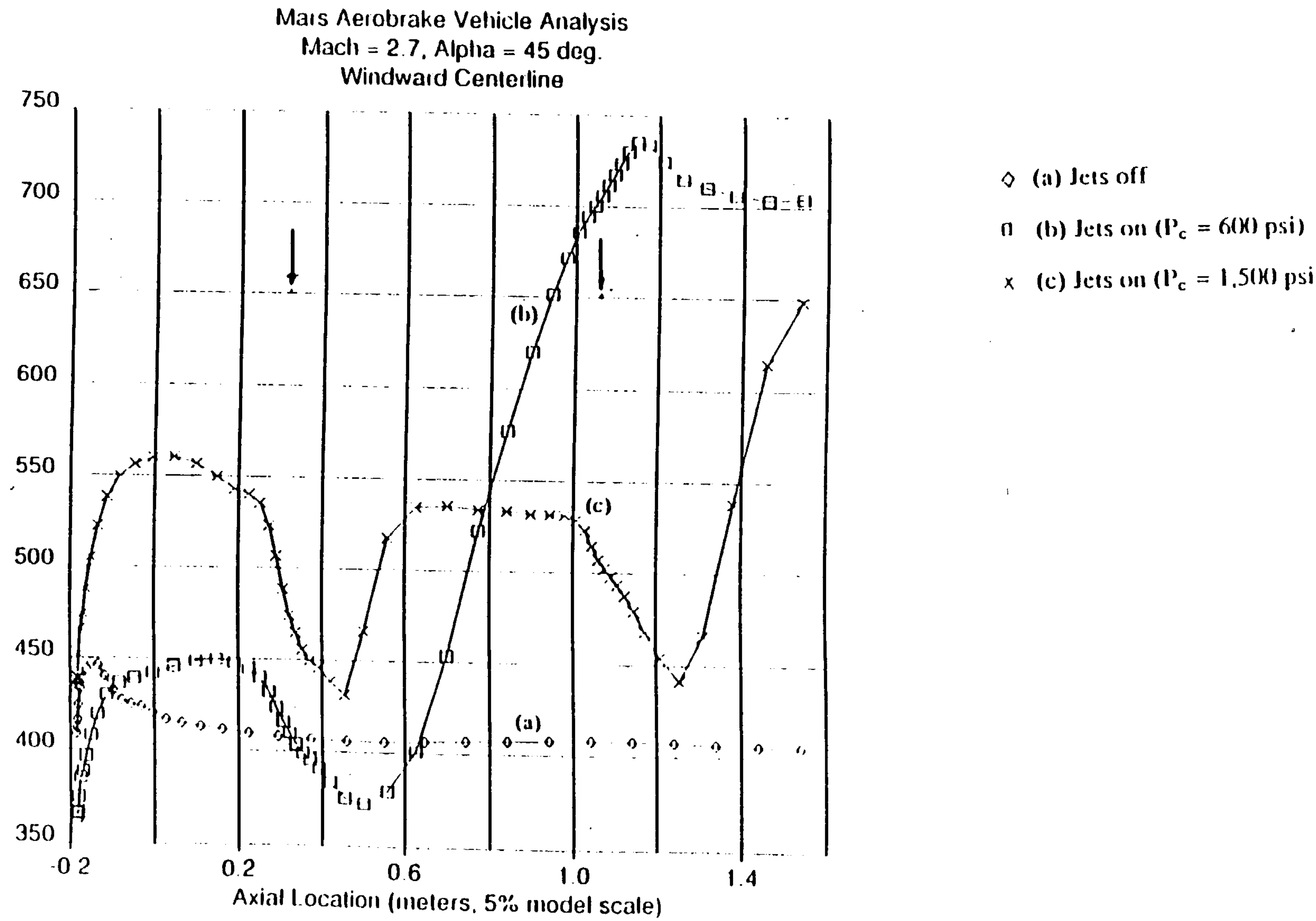
Mars Aerobrake Temperature Distribution



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Jet - Surface Interaction

Physical Model

2 Jets

Jet Velocity	M = 0.16
Jet Exit Width (W)	5.5mm
Distance Between Jets	27.6 W
Height (h) Above Ground Plane	= 60 W, 30 W
Free Stream Velocity	= 0

Reynolds Numbers (based upon jet exit velocity and W) = 16,000

Jet - Surface Interaction

- Numerical Model

  Code Used - Boeing Developed

  Grid Size 111 x 61

  Laminar Navier-Stokes Solution

  Time Accurate, Explicit McCormack Method

  Developed By Cao and Elangovan

  CRAY XMP   1-2 hrs CPU

Jet - Surface Interaction

Preliminary Results

- **Flow Is Unsteady**
- **Oscillating Frequency - 250 Hertz**
- **Unsteady Flow Similarity Observed In Water Table Experiments**

ADVANCED CIVIL SPACE SYSTEMS

Jet - Surface Interaction Velocity Contour

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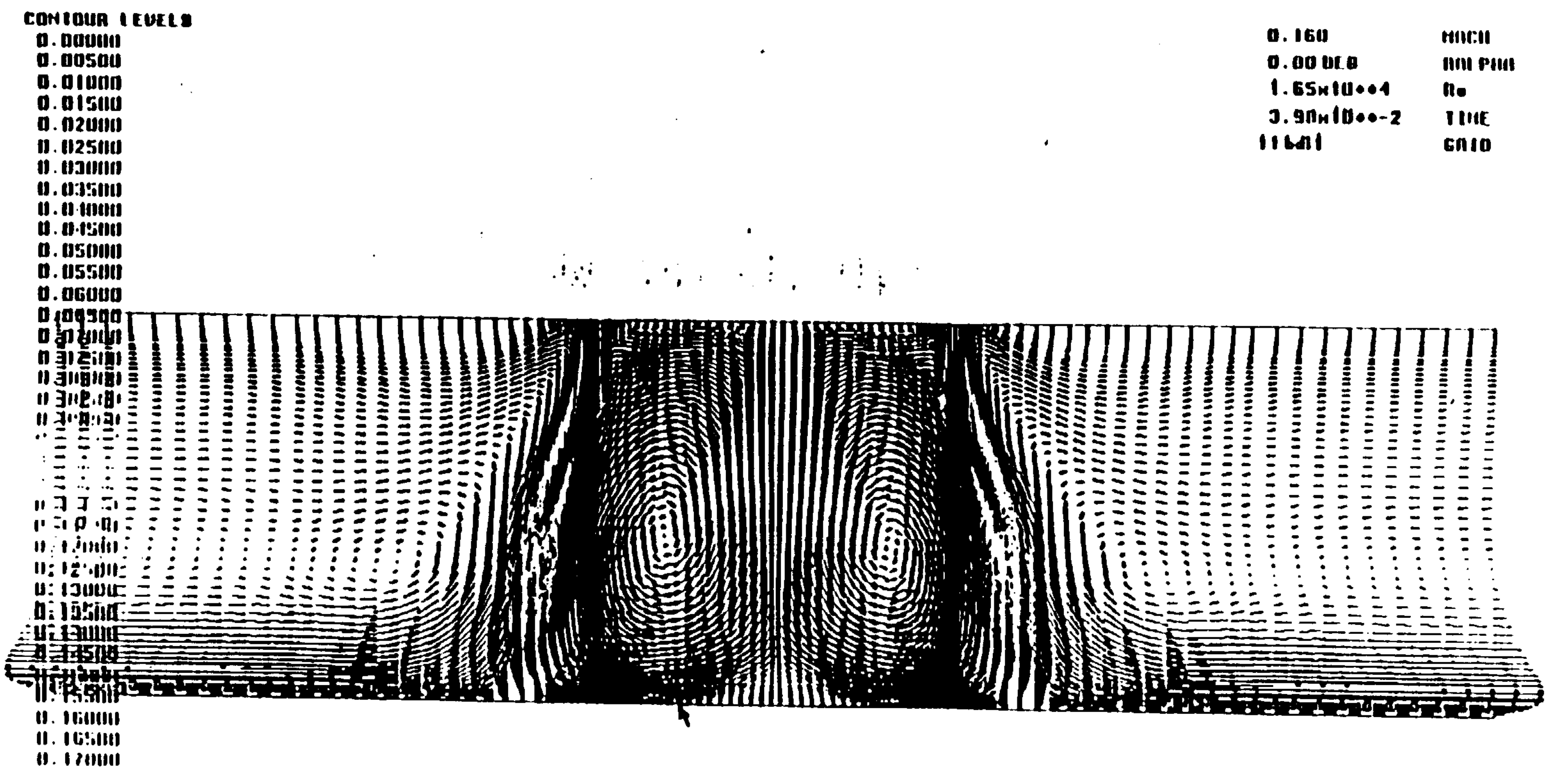

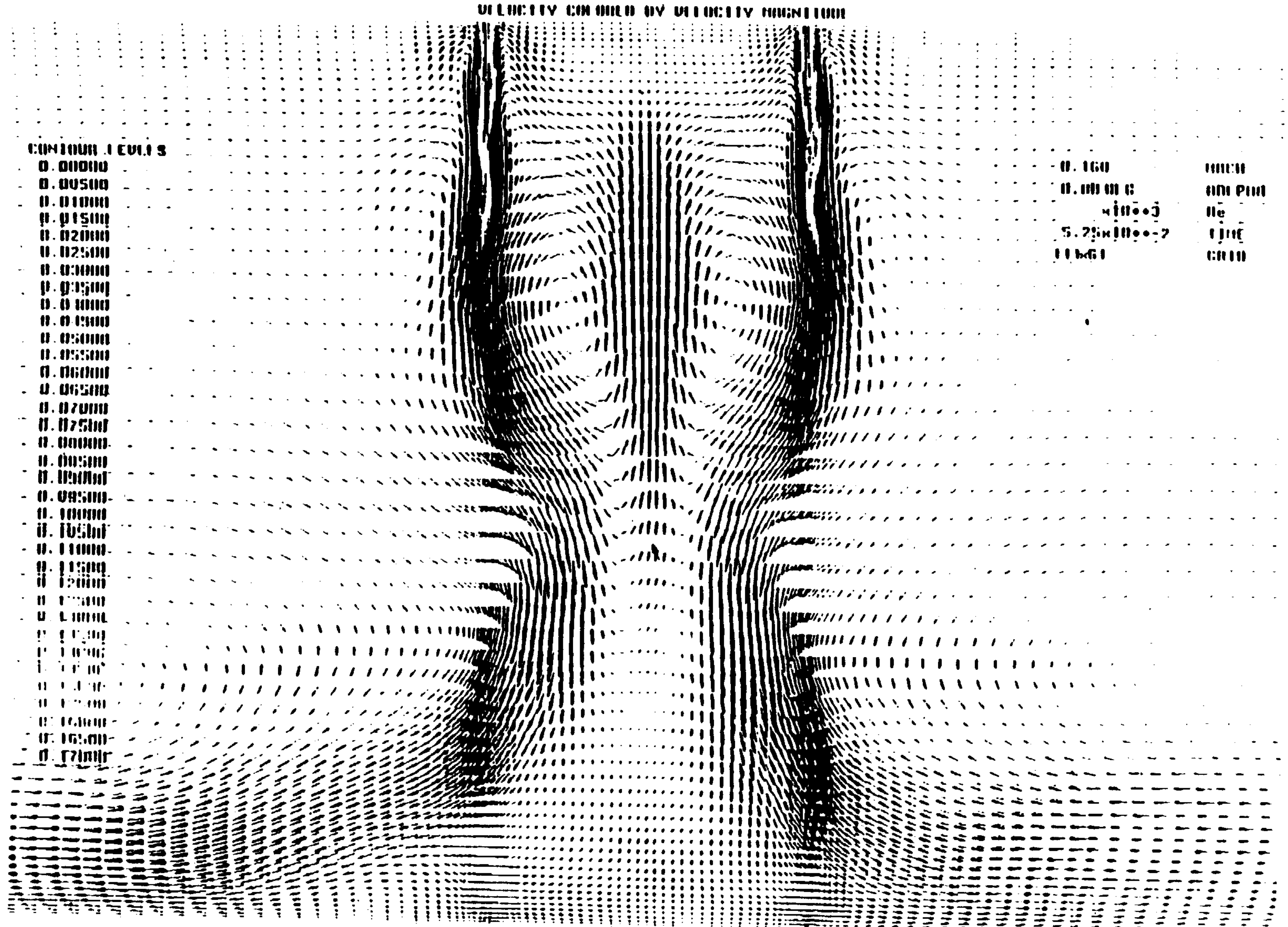
VELOCITY COLORED BY VELOCITY MAGNITUDE
CONTOUR LEVELS
0.00000
0.00500
0.01000
0.01500
0.02000
0.02500
0.03000
0.03500
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0.09000
0.09500
0.10000
0.10500
0.11000
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ADVANCED CIVIL SPACE SYSTEMS

Jet - Surface Interaction Particle Trace

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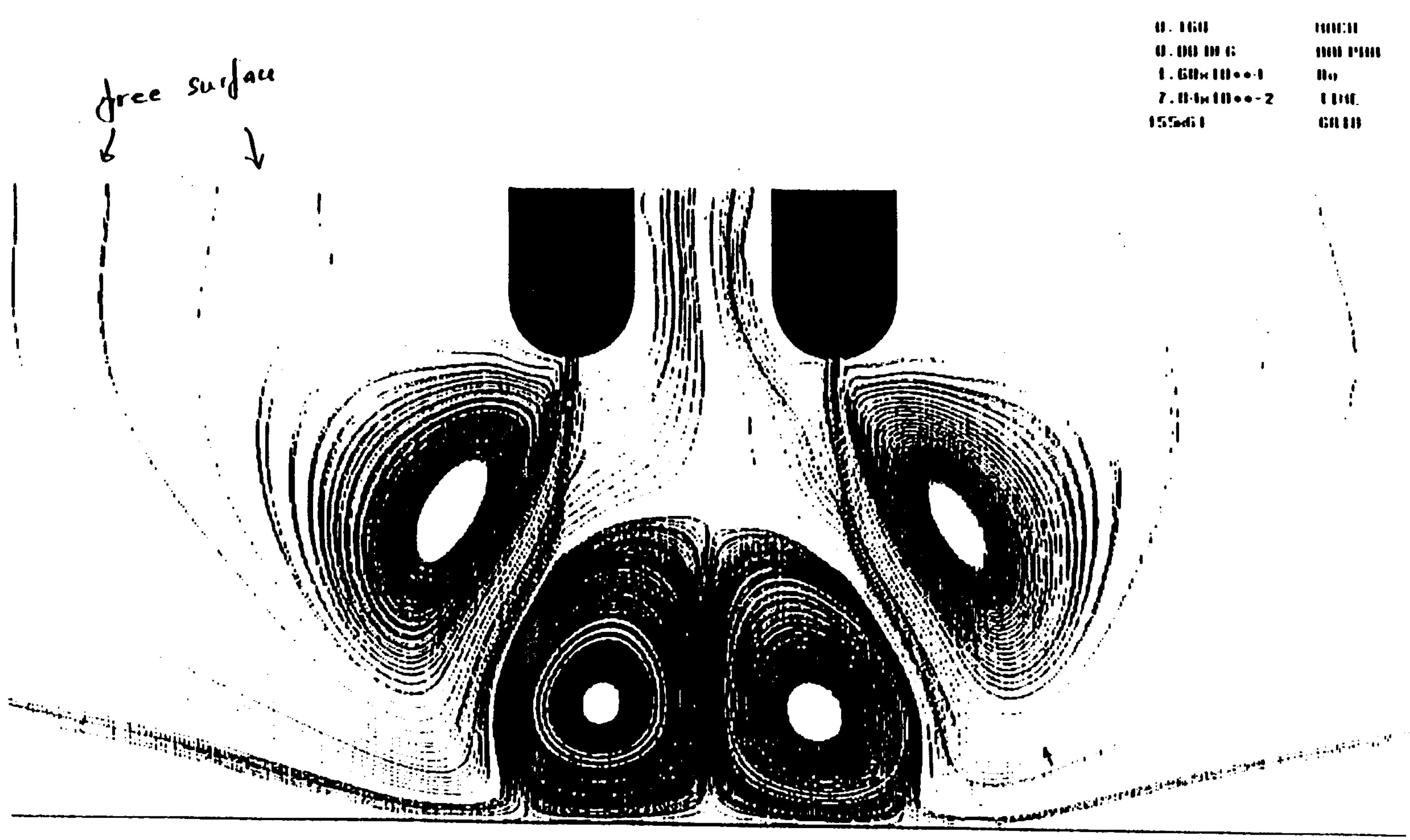
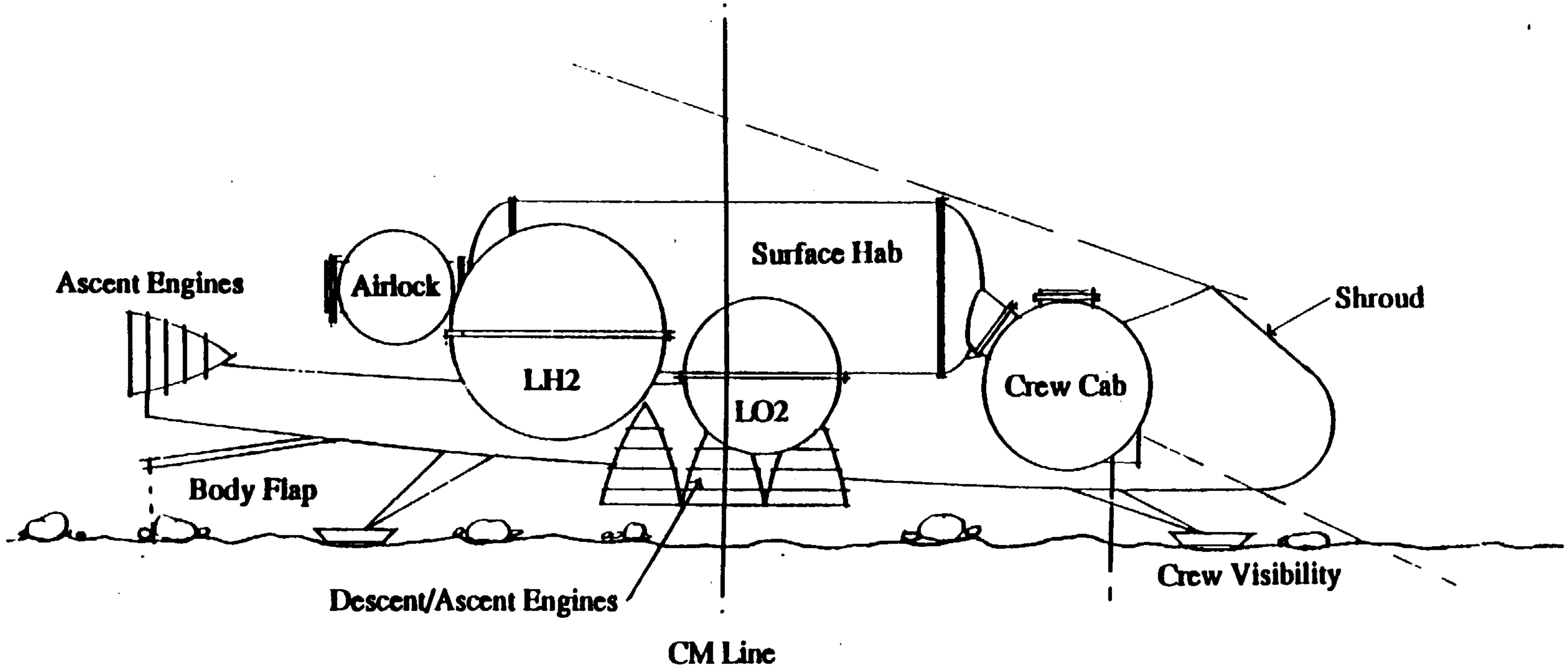


High L/D Reusable MEV Configuration

The high L/D reusable MEV is shown on the facing page. This configuration allows offloading of payload (shown here as a 4.4 m dia. hab module)by way of a ramp and track system, located at the rear of the vehicle. The ramp also acts as a body flap for aero-maneuvering. The crew cab is positioned so that the pilot has down and forward visibility until the landing area is selected. As the vehicle rotates into landing attitude, crew visibility will be limited to the surface directly below.

High L/D (1.1) MEV Configuration

- Payload at CM and removed out back of aerobrake via track system. Back landing legs retract to lower body flap to surface for payload removal
- Crew visibility for landing accommodated through front landing leg door
- Vehicle mass balanced to allow for flight with or without payload



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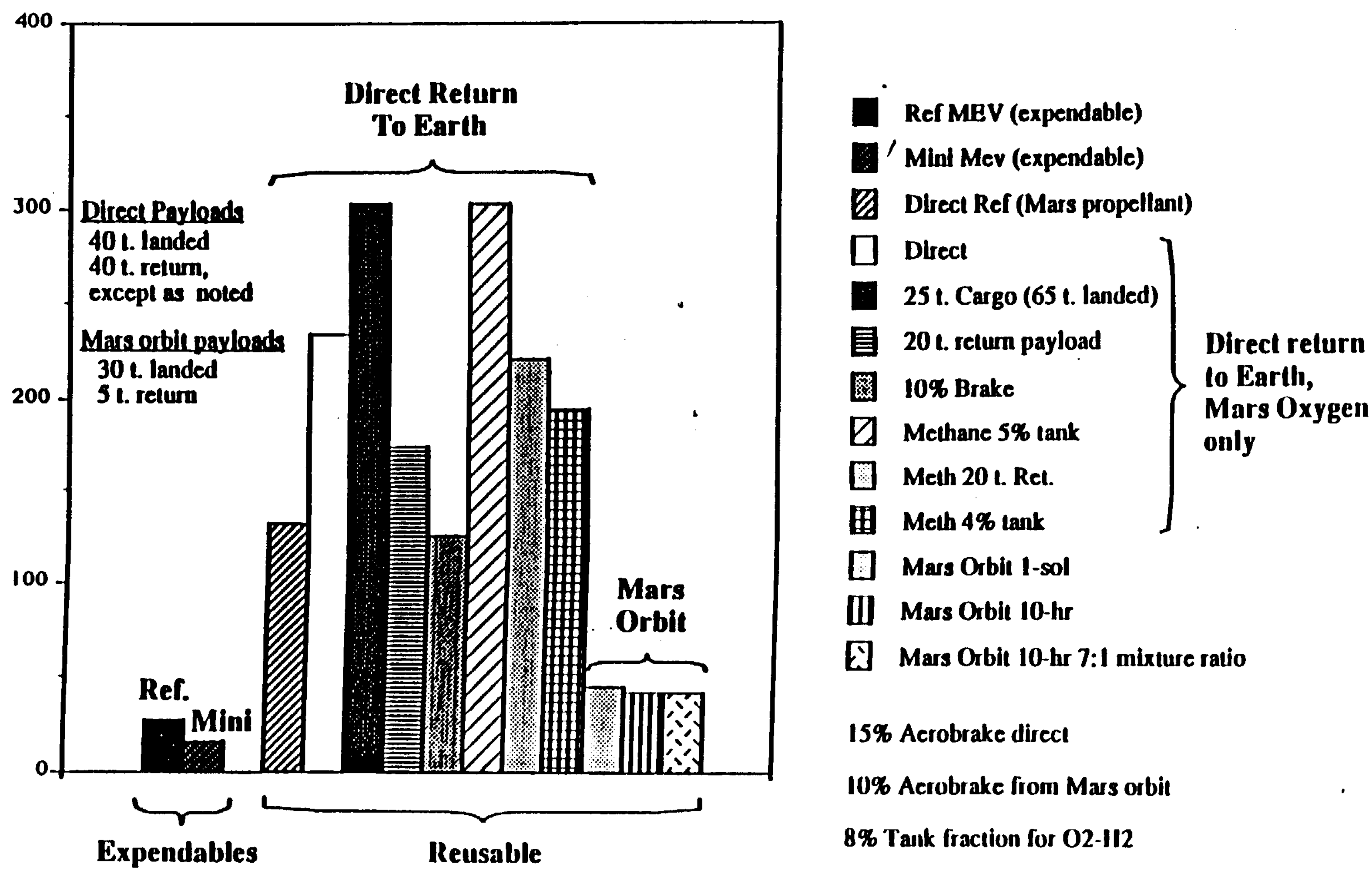
Reusable MEV Sensitivities



15% Aerobrake direct

10% Aerobrake from Mars orbit

8% Tank fraction for O2-H2

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Bi-Conic Lander/Habitat

The Bi-conic lander/habitat is configured to be launched atop a 12m dia. HLLV, and delivered to Mars orbit via a separately launched TMI stage. The Bi-conic shape provides an L/D of 1.3 at a 20 degree angle of attack, and lands using a 6 engine configuration, split 3 forward and 3 aft. The unmanned vehicle is used in an expendable mode, and requires 21 metric tons of propellant for landing.

The 10 crew habitat module delivered to the surface is integrated within the bi-conic, and would not need heavy transportation equipment for deployment. The hab module weighs 40 metric tons when landed, and would need to be outfitted on the surface.

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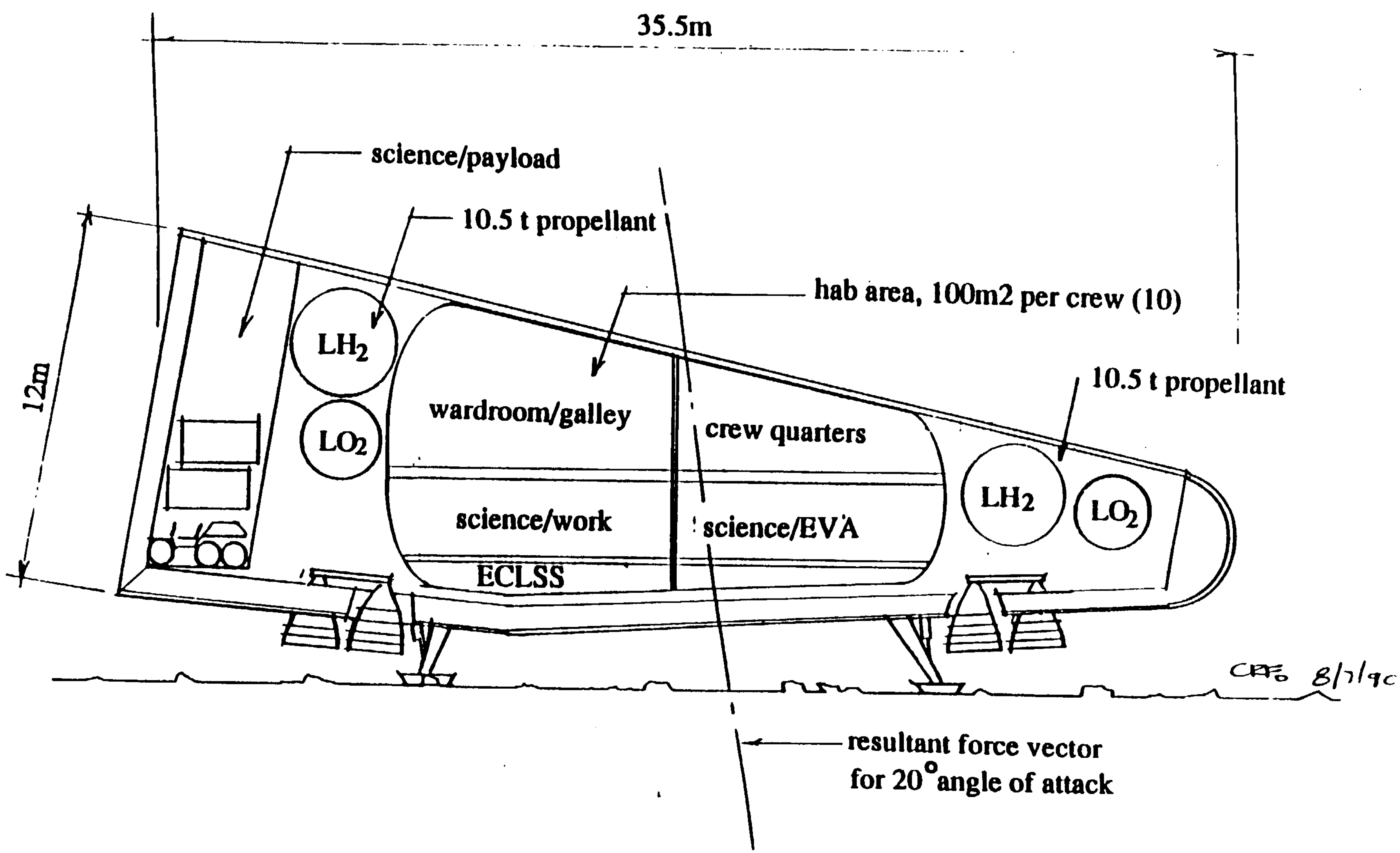
Bi-Conic Lander / Habitat



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REMTECH TECHNICAL NOTE

SUBJECT: Base Heating Environments for the Boeing Mars Hyperboloid Aerobrake (L/D = 0.5)

DATE: January 25, 1991

AUTHORS: Craig P. Schmitz and Carl D. Engel

CONTRACT NO.: HM9611

PREPARED FOR: Boeing Aerospace and Electronics

INTRODUCTION

The intent of this technical note is to document a base heating environment for the Boeing Mars hyperboloid aerobrake (L/D = 0.5). The approach used in this analysis was to develop an empirical method for calculating the convective heating, radiative heating, and wake closure angle.

A side view of the Mars transfer vehicle aerobrake is shown in Fig. 1. The equation of the hyperboloid is provided with this figure. A top view of the MTV aerobrake is provided in Fig. 2.

The trajectory analyzed is an aerocapture trajectory in the Mars atmosphere. The trajectory data provided by Boeing included the time, altitude, velocity, free-stream temperature, free-stream pressure, and free-stream density. The convective, radiative, and total heat rates at stagnation conditions on the front surface of the aerobrake were also provided. The methodologies in this study were developed using the trajectory conditions occurring at the times of highest heating. The trajectory data for the times of peak heating is included in Table 1. In addition an altitude versus velocity plot of this trajectory is provided in Fig. 3.

BASE CONVECTION

An empirical method for calculating the base convective heating is presented in Table 2. Using this method and the geometry and trajectory information provided by Boeing for the MTV a base convective heating environment was calculated and is shown in Fig. 4. The sensitivity of this predicted environment to wall temperature was investigated and is shown in Fig. 5. Note that a 556 K variation in wall temperature resulted in only a 4.8 percent variation in the peak heating rates. The sensitivity to base diameter was also investigated and is shown in Fig. 6. Here it is shown that a 10 percent variation in the base diameter resulted in a 1.8 percent variation in the peak heating rates. In addition to the method selected above for calculating the base convection environment two other independent sources were identified [Ref. 1 & 2] which indicated that base heating rates

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could be approximated as 2 percent of the front face stagnation heating. Both of these reports were for aerobrakes in air. A comparison of the calculated base convective heat rates with the stagnation convective heat rates shows that the base heating approaches 5.6 percent of the stagnation as shown in Fig. 7. The 5.6 percent prediction is calculated using CO_2. The heat transfer coefficient for base convection is higher for CO_2 than for air based solely on the differences in the viscosity and density, which accounts for the major differences in the results of these two methodologies. Another source of error may be the variety of stagnation convective heating methodologies available and used within each of the different references. However, since the methodology recommended in Table 2 is independent of the stagnation heating methodology and is the more conservative of the heating estimates, it is the method for calculating the base convective heating recommended by this study.

BASE RADIATION

An empirical method for calculating the base radiative heating is presented in Table 3. Using this method and the geometry and trajectory information provided by Boeing for the MTV, a base radiative heating environment was calculated and is shown in Fig. 8. The sensitivity of this predicted environment to the base diameter was investigated and is shown in Fig. 9. Here, it is shown that a 10 percent variation in the base diameter resulted in a 5.4 percent variation in the peak heating rates.

WAKE CLOSURE ZONE

The objective was to define the region in the base of the Boeing brake where payloads could be placed with a minimum impact from base heating. The procedure used to estimate this region follows that developed at REMTECH and presented in Ref. [3]. Calculations were performed for the peak forebody heating trajectory time of 114 seconds.

The methodology used to determine the expansion angle of the shear layer, θ_s, is as follows:

1. BLIMPK (Ref. [8]) was used to calculate the boundary layer edge properties for a 100% CO_2 free-stream Martian atmosphere. REMTECH extended the equilibrium property curve fit equation range of applicability from 6,000K to 15,000K in BLIMPK for this study. Results provided in Table 4 indicate that the specific heat ratio at the boundary layer edge is approximately 1.11 and is nearly constant throughout the expansion to the base pressure.
2. The pressure ratio at the forebody sonic point and the sonic point flow angle, θ^* are evaluated using Lee's modified Newtonian pressure relation and an effective post

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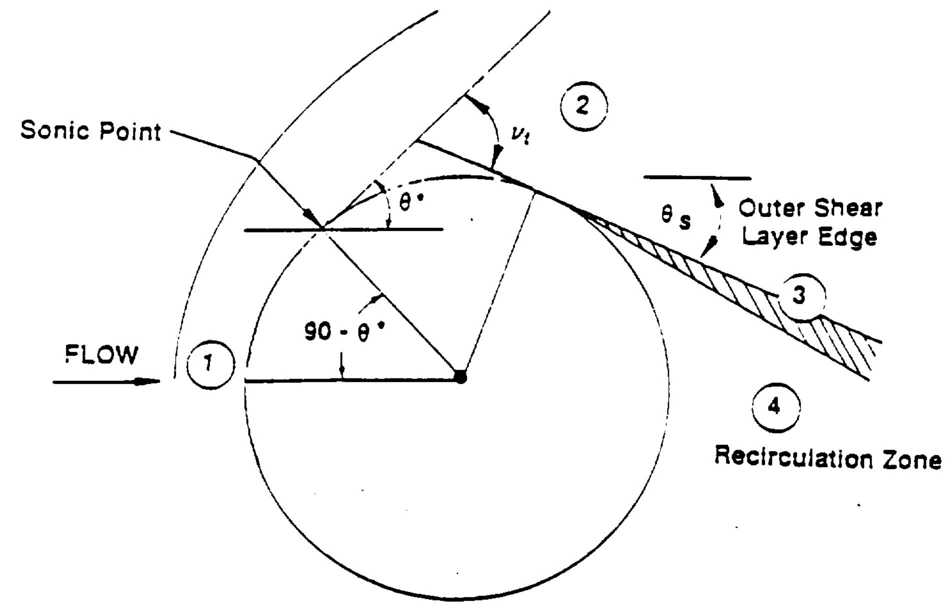
shock γ of 1.11.

$$\frac{P^*}{P_1} = \left(\frac{2}{\gamma+1}\right)^{\frac{\gamma}{\gamma-1}};$$

$$\theta^* = \sin^{-1}\left[\left(\frac{P^*}{P_1}\right)^{1/2}\right] \quad \text{yielding} \quad \theta^* = 49.75^\circ$$



Flow Geometry

3. The nominal base pressure ratio P_4/P_∞ is determined from the relation by Engel (Ref. [4]) for laminar high Reynolds number wakes.

$$\frac{P_4}{P_\infty} = 10^{\left[-0.18708+0.10806(M_\infty-8)^{0.8}\right]}$$

$$\frac{P_4}{P_\infty} = 15.47 \text{ at time} = 114 \text{ sec}$$

It is assumed that the pressure across the shear layer is identical to the pressure in the recirculation zone, hence $P_4/P_1 = P_3/P_1$.

4. P^*/P_3 is computed from steps 2 and 3.
5. Isentropically expanding the flow from P^*, to P_3, (Prandtl-Meyer expansion theory), the Mach No. (M_3) of the expanded local flow is determined.

$$M_3 = \left[\left(\frac{\gamma+1}{\gamma-1}\right)\left(\frac{P^*}{P_3}\right)^{\frac{\gamma-1}{\gamma}} - \left(\frac{2}{\gamma-1}\right)\right]^{1/2}$$

6. If the turning angle of the outer boundary of the shear layer is defined as ν_t, then it is determined from

$$\nu_t = \sqrt{\frac{\gamma+1}{\gamma-1}}\tan^{-1}\left[\sqrt{\left(\frac{\gamma-1}{\gamma+1}\right)(M_3^2-1)}\right] - \tan^{-1}\left(\sqrt{M_3^2-1}\right)$$

7. The angle of the outer edge of the shear layer relative to the free-stream velocity vector is then

$$\theta_s = \nu_t - \theta^*$$
$$\theta_s = 81.84 - 49.75$$
$$= 32.09 \qquad \text{for } \gamma = 1.11$$

Based on the results from the preceding procedure, the wake closure zone shown in Fig. 10 was defined. The angle, θ_s, is the outer shear layer angle. The low heating region in the wake occurs on the inside of the shear layer. Several potential methods could be used to define this inner edge (heating, velocity or shear level). In the current quick analysis the shear layer spreading angle was measured off of a schlieren of the AFE presented in Ref. [5]. This data was for air at Mach 6. The measured shear layer angle was 7 degrees.

In order to understand the flowfield around the blunt end of the brake, the plot in Fig. 11 was prepared. The flow properties shown are from the Table 4 BLIMPK results plotted on the surface of the brake. The straight lines issuing from the corner are Mach waves calculate from boundary layer edge conditions. The shock shape was derived from AFE schlieren presented in Ref. [6] for CF_4 at Mach 6 (ρ_1/ρ_∞ = 11.7). The wind tunnel shock shape was adjusted to account for shock density ratio effects using

$\delta/R = 0.667/(\rho_1/\rho_\infty - 1)$ Nondimensional stand-off distance

where $\rho_1/\rho_\infty = 14.73$ for time 114 sec in CO_2

$\rho_1/\rho_\infty = 11.7$ for CF_4(tunnel)

yielding $\dfrac{\delta_{flt}}{\delta_{tun}} = 0.779$

The Mach wave was drawn at the Mach angle at the body to the shock. Compatibility of these conditions at the shock were not verified. More than likely the Mach waves bend to the right when moving along the Mach line from the body to the shock.

BRAKE DESIGN CONSIDERATIONS

During the analysis process of determining the brake wake heating environment, several observations regarding the current brake design were made. These observations are:

1. The geometric stagnation point is located quite near the blunt side of the brake at 20 degrees angle of attack

2. Expansion from the geometric stagnation point along the hyperboloid is only about 11 degrees.
3. As a result of (2) the sonic point is located on the torus (See Fig. 11).
4. Although specific calculations were not performed, it is anticipated that the constraint of the sonic point to the torus region will produce a high heating region on the hyperboloid. This high heating region would probably be higher convectively than the stagnation region by a factor of 1.2 to 1.4. This is a similar flow situation to that measured by Ref. [7] as shown in Fig. 11. The bluntness, X^*/R^*, of the upper portion of the brake is about 0.144.
5. The effective radius based on this bluntness ($X^*/R^* = 0.144$) is approximately 13.65 meters. This is nearly the same as the effective radius of 13.0 meters used in both the forebody convective and radiative heating calculations.
6. The constrained sonic point produces higher than Newtonian pressures between the stagnation and sonic point. Consequently, aerodynamics based on Newtonian theory alone are inadequate for this type of body.

Based on the preceding observations and work presented in preceding section the following recommendations are made:

1. New brake geometries or modifications of the current brake geometry as shown in Fig. 12 should be examined. The hyperboloid or primary blunt surface should be designed to accommodate the sonic point which occurs at a turning angle of approximately 40 degrees from the stagnation point. If large excursions (above 5 degrees) in angle of attack are anticipated, this should be accounted for as well. The extended brake as shown in Fig. 12 would also substantially increase the usable wake payload volume.
2. The torus radius should be held constant for design simplicity. If the sonic point is not located on the torus, the heating will be more benign and lower environments will exist. The torus radius environment can be determined using BLIMPK (Ref. [8] for convection and RADCOR (Ref. [9]) for radiation.
3. The new and modified brake face and wake environment should be examined at three altitudes using BLIMPK boundary layer calculations, LANMIN (Ref. [10]) for pressure distributions, RADCOR for radiation distributions and correlations presented herein for wake conditions. By examining the brake geometry at three separated Reynolds numbers all potential heating design considerations will be identified.
4. Computational fluid dynamics code validation calculations should follow basic design trades preformed using engineering codes.

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Table 1: MTV Trajectory Data

Time (Sec)	Alt. (KM)	Vel. (KM/sec)	Temp. (K)	Press. (N/M^2)	Density (KG/M^3)	QC (W/CM^2)	QR (W/CM^2)	QTOT (W/CM^2)
110	43.7	6.958	172.03	11.51864	0.3678D-03	28.66	56.92	85.58
112	43.1	6.913	172.60	12.29919	0.3914D-03	29.08	57.73	86.81
114	42.6	6.866	173.14	13.06595	0.4145D-03	29.40	57.81	87.21
116	42.1	6.817	173.63	13.81158	0.4369D-03	29.64	57.09	86.73
118	41.6	6.766	174.08	14.52953	0.4584D-03	29.78	55.55	85.33
120	41.2	6.714	174.49	15.21327	0.4788D-03	29.84	53.17	83.01
122	40.8	6.660	174.86	15.85678	0.4980D-03	29.81	50.64	80.46
124	40.5	6.604	175.19	16.45464	0.5158D-03	29.71	47.65	77.35
126	40.2	6.548	175.49	17.00229	0.5321D-03	29.53	43.98	73.51
128	39.9	6.491	175.74	17.49571	0.5468D-03	29.27	39.83	69.10
130	39.7	6.433	175.96	17.93181	0.5597D-03	28.96	35.76	64.72
132	39.5	6.375	176.15	18.30809	0.5708D-03	28.59	31.26	59.85
134	39.3	6.316	176.30	18.62283	0.5801D-03	28.16	26.41	54.58
136	39.2	6.258	176.42	18.87512	0.5876D-03	27.70	21.83	49.53
138	39.1	6.201	176.52	19.06487	0.5932D-03	27.19	17.28	44.47
140	39.0	6.143	176.58	19.19251	0.5970D-03	26.65	12.72	39.37
142	39.0	6.087	176.61	19.25920	0.5989D-03	26.08	8.76	34.85
144	39.0	6.031	176.61	19.26657	0.5992D-03	25.49	4.83	30.32
146	39.0	5.976	176.59	19.21661	0.5977D-03	24.89	0.00	24.89
148	39.1	5.922	176.54	19.11180	0.5946D-03	24.27	0.00	24.27
150	39.2	5.870	176.46	18.95515	0.5900D-03	23.65	0.00	23.65

Table 2: Base Convection Heating

The procedure for calculating convective heating in the base region follows that of Warmbrod. This method can be used for both air and CO_2 atmospheres. References to curve fit equations for calculating viscosities have been included for use with a Martian atmosphere assumed to be 100%CO_2.

Base Pressure: (Engel)

$$P_B = P_\infty \, 10^{(-.18708 + .10806\,(M_\infty - 8.)^{.8})} \qquad \text{(lbf/ft}^2\text{)}$$

where

P_∞ = free-stream pressure (lbf/ft^2)
M_∞ = free-stream Mach number (dimensionless)

Free-stream Mach Number:

$$M_\infty = \frac{u_\infty}{\sqrt{g\gamma R T_\infty}} \qquad \text{(Dimensionless)}$$

where

u_∞ = free-stream velocity (ft/s)
g = 32.174 (lbm ft/lbf s^2)
γ = 1.40 for CO_2 at free-stream conditions (Dimensionless)
R = 35.10 (ft lbf/lbm R)
T_∞ = free-stream temperature (R)

Base Enthalpy: (Bulmer)

$$H_B = H_s\left(.26 + .651\left(\frac{H_W}{H_S}\right)\right) \qquad \text{(BTU/lbm)}$$

where

H_S = stagnation enthalpy = $\frac{u_\infty^2}{50073.8} + C_p T_\infty$ (BTU/lbm)
H_W = wall enthalpy (BTU/lbm)
C_p = specific heat of CO_2 (BTU/lbm R)

Viscosity:

For $P < .0001$ atm use curve fit of Candler
For $P \geq .0001$ atm use curve fit of Marvin & Deiwert

Free-stream Reynolds number:

$$(Re_\infty)_D = \frac{\rho_\infty u_\infty D}{\mu_\infty} \qquad \text{(Dimensionless)}$$

where

ρ_∞ = free-stream density (slugs/ft^3)
D = base diameter (ft)
μ_∞ = free-stream viscosity (lb/ft-s)

Table 2: (Continued) Base Convection Heating

Heat Transfer Coefficient: (Warmbrod)

$$h_B = .349\ \rho_\infty\ u_\infty\ (Re_\infty)_D^{-.1722} \left(\frac{\mu_\infty}{\mu_B}\right)^{-.1722} \qquad (BTU/ft^2\text{-}s\text{-}^\circ R)$$

where

$$\mu_B = \text{base viscosity (lb/ft-s)}$$

Base Convection Heat Rate:

$$\dot{q}_B = h_B\ (H_B - H_W) \qquad (BTU/ft^2\text{-}s)$$

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Table 3: Base Radiation Heating

The procedure for calculating radiation heating in the base region follows that developed by Hearne. The following empirical equation was derived to correlate experimental wake radiation data measured in the NASA-Ames Research Center free-flight hypersonic facility. Measurements were made over a range of velocities from 20,000 ft/sec to 32,000 ft/sec for several ablation materials.

$$\dot{q}_{RAD} = C\sqrt{D}\ \rho_\infty \left(\frac{u_\infty}{10^4}\right)^7 \cos\ \theta \qquad (BTU/ft^2 sec)$$

where

C = constant of proportionality — ablation material dependent
D = vehicle diameter (ft)
ρ_∞ = free-stream density (lbm/ft^3)
u_∞ = free-stream velocity (ft/s)
θ = angle between the local surface normal and the wake angle

A check calculation of this relation was made for air using project FIRE I flight data. The results were

$$\begin{array}{ll} q_{rad} = 1.33 & \text{Btu/sft-sec (calculated)} \\ q_{rad} = 1.41 & \text{Btu/sft-sec (measured)} \end{array} \quad @ \quad \begin{cases} \rho_\infty = 4.745 \times 10^{-6} & \text{lbm/cft} \\ u_\infty = 37{,}840 & \text{ft/sec} \\ D = 2.204 & \text{ft} \end{cases}$$

A value of 17.0 ($BTU\text{-}s^6$)/($lbm\text{-}ft^{6.5}$) for the constant C was obtained using Lexan as an ablator. This selection of ablation material resulted in the highest radiation heating in the wake region and was recommended in application of the above equation.

This correlation when applied to a range of θ from 0 to 90, calculates radiative heating rates which are at a maximum at $\theta = 0$ and decrease to a 0 heat rate at $\theta = 90$. For the AFE aerobraking vehicle the minimum wake radiation heating rates were calculated to be 36.6 percent of the peak radiation heat rates as determined by Sambamurthi. From this relationship the following correlation was derived for estimating the radiation heating rates in the base for Mars atmospheres.

$$\dot{q}_{RAD} = C\sqrt{D}\ \rho_\infty \left(\frac{u_\infty}{10^4}\right)^7 (.634 \cos\theta + .366) \qquad (BTU/ft^2 sec)$$

REFERENCES

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Table 4: BLIMPK Boundary Layer Edge Conditions for Time = 114 sec

```
STATION NO    1

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
         0.34931E+00 0.46617E+01-0.19159E+01 0.94477E-01 0.11210E+01

      TEMP= 6643.0844 DEG-K      PRES= 0.1858 ATM      MOL WT= 17.9030978
       ENTHALPY= 0.5530450E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
       DENSITY= 0.380900E-03 LB/CUFT
    VEL= 0.000E+00 FT/SEC    MACH= 0.000E+00    AREA= 0.000E+00 SOFT/LB/SEC

    SPECIES   MOLE FR.         SPECIES   MOLE FR.        SPECIES   MOLE FR.
          C  0.18621E+00             O  0.59311E+00              O2 0.44045E-04

         CO 0.22063E+00           CO2 0.53997E-05


STATION NO    2

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
         0.34931E+00 0.46618E+01-0.19159E+01 0.94477E-01 0.11210E+01

      TEMP= 6642.9440 DEG-K      PRES= 0.1857 ATM      MOL WT= 17.9033077
       ENTHALPY= 0.5530226E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
       DENSITY= 0.380796E-03 LB/CUFT
    VEL= 0.142E+03 FT/SEC    MACH= 0.233E-01    AREA= 0.185E+02 SOFT/LB/SEC

    SPECIES   MOLE FR.         SPECIES   MOLE FR.        SPECIES   MOLE FR.
          C  0.18620E+00             O  0.59310E+00              O2 0.44043E-04
         CO 0.22065E+00           CO2 0.53994E-05


STATION NO    3

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
         0.34930E+00 0.46622E+01-0.19160E+01 0.94477E-01 0.11210E+01

      TEMP= 6642.5228 DEG-K      PRES= 0.1856 ATM      MOL WT= 17.9039376
       ENTHALPY= 0.5529552E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
       DENSITY= 0.380486E-03 LB/CUFT
    VEL= 0.284E+03 FT/SEC    MACH= 0.466E-01    AREA= 0.924E+01 SOFT/LB/SEC

    SPECIES   MOLE FR.         SPECIES   MOLE FR.        SPECIES   MOLE FR.
          C  0.18617E+00             O  0.59309E+00              O2 0.44027E-04
         CO 0.22069E+00           CO2 0.53987E-05


STATION NO    4

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
         0.34926E+00 0.46648E+01-0.19168E+01 0.94477E-01 0.11209E+01

      TEMP= 6639.5723 DEG-K      PRES= 0.1844 ATM      MOL WT= 17.9083524
       ENTHALPY= 0.5524834E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
       DENSITY= 0.378319E-03 LB/CUFT
    VEL= 0.711E+03 FT/SEC    MACH= 0.117E+00    AREA= 0.372E+01 SOFT/LB/SEC

    SPECIES   MOLE FR.         SPECIES   MOLE FR.        SPECIES   MOLE FR.
          C  0.18597E+00             O  0.59299E+00              O2 0.43913E-04
         CO 0.22099E+00           CO2 0.53931E-05


STATION NO    5

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
         0.34912E+00 0.46740E+01-0.19198E+01 0.94478E-01 0.11207E+01
```

Table 4: (Continued) BLIMPK Boundary Layer Edge Conditions for Time = 114 sec

```
        TEMP= 6629.0072 DEG-K       PRES=  0.1802 ATM      MOL WT= 17.9241934
         ENTHALPY= 0.5507930E+04 CAL/GM       ENTROPY= 0.35512E+01 CAL/GM-DEG K
         DENSITY= 0.370643E-03 LB/CUFT
    VEL= 0.142E+04 FT/SEC    MACH= 0.234E+00    AREA= 0.189E+01 SQFT/LB/SEC

    SPECIES   MOLE FR.        SPECIES   MOLE FR.        SPECIES   MOLE FR.
         C  0.18525E+00             O  0.59263E+00            O2 0.43506E-04
         CO 0.22207E+00           CO2 0.53733E-05

  STATION NO    6

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP      GAMMA
         0.34888E+00 0.46892E-01-0.19246E+01 0.94471E-01 0.11203E+01

        TEMP= 6611.2981 DEG-K       PRES=  0.1734 ATM      MOL WT= 17.9508596
         ENTHALPY= 0.5479558E+04 CAL/GM       ENTROPY= 0.35512E+01 CAL/GM-DEG K
         DENSITY= 0.358063E-03 LB/CUFT
    VEL= 0.214E+04 FT/SEC    MACH= 0.352E+00    AREA= 0.130E+01 SQFT/LB/SEC

    SPECIES   MOLE FR.        SPECIES   MOLE FR.        SPECIES   MOLE FR.
         C  0.18404E+00             O  0.59202E+00            O2 0.42829E-04
         CO 0.22389E+00           CO2 0.53397E-05

  STATION NO    7

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP      GAMMA
         0.34854E+00 0.47100E-01-0.19310E+01 0.94445E-01 0.11198E+01

        TEMP= 6586.2862 DEG-K       PRES=  0.1641 ATM      MOL WT= 17.9887650
         ENTHALPY= 0.5439410E+04 CAL/GM       ENTROPY= 0.35512E+01 CAL/GM-DEG K
         DENSITY= 0.340898E-03 LB/CUFT
    VEL= 0.286E+04 FT/SEC    MACH= 0.473E+00    AREA= 0.102E+01 SQFT/LB/SEC

    SPECIES   MOLE FR.        SPECIES   MOLE FR.        SPECIES   MOLE FR.
         C  0.18231E+00             O  0.59116E+00            O2 0.41884E-04
         CO 0.22647E+00           CO2 0.52914E-05

  STATION NO    8

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP      GAMMA
         0.34810E+00 0.47361E-01-0.19389E+01 0.94382E-01 0.11192E+01

        TEMP= 6553.7338 DEG-K       PRES=  0.1526 ATM      MOL WT= 18.0385247
         ENTHALPY= 0.5387025E+04 CAL/GM       ENTROPY= 0.35512E+01 CAL/GM-DEG K
         DENSITY= 0.319581E-03 LB/CUFT
    VEL= 0.359E+04 FT/SEC    MACH= 0.596E+00    AREA= 0.870E+00 SQFT/LB/SEC

    SPECIES   MOLE FR.        SPECIES   MOLE FR.        SPECIES   MOLE FR.
         C  0.18005E+00             O  0.59003E+00            O2 0.40672E-04
         CO 0.22987E+00           CO2 0.52274E-05
```

Table 4: (Continued) BLIMPK Boundary Layer Edge Conditions for Time = 114 sec

```
STATION NO   9

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP      GAMMA
         0.34756E+00 0.47666E+01-0.19478E+01 0.94256E-01 0.11184E+01

       TEMP= 6513.3024 DEG-K       PRES=  0.1394 ATM       MOL WT= 18.1010011
        ENTHALPY= 0.5321758E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
        DENSITY= 0.294656E-03 LB/CUFT
   VEL= 0.434E+04 FT/SEC    MACH= 0.722E+00    AREA= 0.783E+00 SQFT/LB/SEC

   SPECIES    MOLE FR.        SPECIES    MOLE FR.        SPECIES    MOLE FR.
         C  0.17721E+00             O  0.58862E+00              O2 0.39197E-04
         CO 0.23413E+00           CO2 0.51459E-05

STATION NO  10

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP      GAMMA
         0.34612E+00 0.48369E+01-0.19665E+01 0.93669E-01 0.11163E+01

       TEMP= 6406.7744 DEG-K       PRES=  0.1091 ATM       MOL WT= 18.2591960
        ENTHALPY= 0.5148772E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
        DENSITY= 0.236620E-03 LB/CUFT
   VEL= 0.586E+04 FT/SEC    MACH= 0.991E+00    AREA= 0.721E+00 SQFT/LB/SEC

   SPECIES    MOLE FR.        SPECIES    MOLE FR.        SPECIES    MOLE FR.
         C  0.16956E+00             O  0.58480E+00              O2 0.35472E-04
         CO 0.24560E+00           CO2 0.49228E-05

STATION NO  11

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP      GAMMA
         0.34465E+00 0.48912E+01-0.19777E+01 0.92677E-01 0.11143E+01

       TEMP= 6298.7872 DEG-K       PRES=  0.0844 ATM       MOL WT= 18.4450081
        ENTHALPY= 0.4972030E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
        DENSITY= 0.187914E-03 LB/CUFT
   VEL= 0.709E+04 FT/SEC    MACH= 0.122E+01    AREA= 0.750E+00 SQFT/LB/SEC

   SPECIES    MOLE FR.        SPECIES    MOLE FR.        SPECIES    MOLE FR.
         C  0.16156E+00             O  0.58081E+00              O2 0.31942E-04
         CO 0.25759E+00           CO2 0.46864E-05

STATION NO  12

         CP-FROZEN     CP-EQUIL    DLNM/DLNT    DLNM/DLNP      GAMMA
         0.34357E+00 0.49192E+01-0.19803E+01 0.91664E-01 0.11129E+01

       TEMP= 6218.3775 DEG-K       PRES=  0.0693 ATM       MOL WT= 18.5793942
        ENTHALPY= 0.4839620E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
```

Table 4: (Continued) BLIMPK Boundary Layer Edge Conditions for Time = 114 sec

```
          DENSITY= 0.157428E-03 LB/CUFT
     VEL= 0.789E+04 FT/SEC    MACH= 0.137E+01     AREA= 0.805E+00 SQFT/LB/SEC

     SPECIES   MOLE FR.         SPECIES   MOLE FR.         SPECIES   MOLE FR.
           C  0.15545E+00             O  0.57775E+00              O2 0.29476E-04
          CO  0.26676E+00           CO2  0.45055E-05

 STATION NO  13

          CP-FROZEN    CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
          0.34256E+00 0.49347E+01-0.19781E+01 0.90515E-01 0.11116E+01

        TEMP= 6144.2547 DEG-K       PRES=  0.0575 ATM      MOL WT= 18.7058860
        ENTHALPY= 0.4717031E+04 CAL/GM       ENTROPY= 0.35512E+01 CAL/GM-DEG K
        DENSITY= 0.133175E-03 LB/CUFT
     VEL= 0.856E+04 FT/SEC    MACH= 0.150E+01     AREA= 0.877E+00 SQFT/LB/SEC

     SPECIES   MOLE FR.         SPECIES   MOLE FR.         SPECIES   MOLE FR.
           C  0.14970E+00             O  0.57488E+00              O2 0.27326E-04
          CO  0.27539E+00           CO2  0.43365E-05

 STATION NO  14

          CP-FROZEN    CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
          0.34152E+00 0.49393E+01-0.19706E+01 0.89080E-01 0.11103E+01

        TEMP= 6066.5596 DEG-K       PRES=  0.0471 ATM      MOL WT= 18.8411412
        ENTHALPY= 0.4588062E+04 CAL/GM       ENTROPY= 0.35512E+01 CAL/GM-DEG K
        DENSITY= 0.111274E-03 LB/CUFT
     VEL= 0.922E+04 FT/SEC    MACH= 0.163E+01     AREA= 0.975E+00 SQFT/LB/SEC

     SPECIES   MOLE FR.         SPECIES   MOLE FR.         SPECIES   MOLE FR.
           C  0.14355E+00             O  0.57181E+00              O2 0.25199E-04
          CO  0.28461E+00           CO2  0.41581E-05

 STATION NO  15

          CP-FROZEN    CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
          0.33965E+00 0.49507E+01-0.19568E+01 0.86698E-01 0.11082E+01

        TEMP= 5938.8272 DEG-K       PRES=  0.0384 ATM      MOL WT= 19.0318266
        ENTHALPY= 0.4402900E+04 CAL/GM       ENTROPY= 0.35512E+01 CAL/GM-DEG K
        DENSITY= 0.936661E-04 LB/CUFT
     VEL= 0.101E+05 FT/SEC    MACH= 0.181E+01     AREA= 0.106E+01 SQFT/LB/SEC

     SPECIES   MOLE FR.         SPECIES   MOLE FR.         SPECIES   MOLE FR.
           C  0.13487E+00             O  0.56748E+00              O2 0.18217E-04
          CO  0.29762E+00           CO2  0.37456E-05
 STATION NO  16

          CP-FROZEN    CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
          0.33862E+00 0.49237E+01-0.19360E+01 0.84734E-01 0.11071E+01

        TEMP= 5863.8002 DEG-K       PRES=  0.0313 ATM      MOL WT= 19.1692041
        ENTHALPY= 0.4277170E+04 CAL/GM       ENTROPY= 0.35512E+01 CAL/GM-DEG K
        DENSITY= 0.778533E-04 LB/CUFT
     VEL= 0.106E+05 FT/SEC    MACH= 0.193E+01     AREA= 0.121E+01 SQFT/LB/SEC

     SPECIES   MOLE FR.         SPECIES   MOLE FR.         SPECIES   MOLE FR.
           C  0.12863E+00             O  0.56436E+00              O2 0.16778E-04
          CO  0.30699E+00           CO2  0.35811E-05
```

Table 4: (Continued) BLIMPK Boundary Layer Edge Conditions for Time = 114 sec

```
STATION NO  17

        CP-FROZEN    CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
        0.33752E+00 0.48783E-01-0.19068E+01 0.82354E-01 0.11059E+01

      TEMP= 5783.3019 DEG-K       PRES=  0.0250 ATM      MOL WT= 19.3192452
       ENTHALPY= 0.4142112E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
       DENSITY= 0.635494E-04 LB/CUFT
   VEL= 0.112E+05 FT/SEC   MACH= 0.205E+01    AREA= 0.141E+01 SQFT/LB/SEC

   SPECIES   MOLE FR.        SPECIES   MOLE FR.       SPECIES   MOLE FR.
        C  0.12181E+00             O  0.56095E+00            O2 0.15341E-04
        CO 0.31722E+00           CO2 0.34073E-05
STATION NO  18

        CP-FROZEN    CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
        0.33645E+00 0.48162E-01-0.18708E+01 0.79741E-01 0.11049E+01

      TEMP= 5704.4673 DEG-K       PRES=  0.0200 ATM      MOL WT= 19.4687181
       ENTHALPY= 0.4009819E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
       DENSITY= 0.518541E-04 LB/CUFT
   VEL= 0.117E+05 FT/SEC   MACH= 0.217E+01    AREA= 0.165E+01 SQFT/LB/SEC

   SPECIES   MOLE FR.        SPECIES   MOLE FR.       SPECIES   MOLE FR.
        C  0.11501E+00             O  0.55756E+00            O2 0.14038E-04
        CO 0.32741E+00           CO2 0.32409E-05
STATION NO  19

        CP-FROZEN    CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
        0.33539E+00 0.47343E+01-0.18266E+01 0.76805E-01 0.11038E+01

      TEMP= 5624.6137 DEG-K       PRES=  0.0158 ATM      MOL WT= 19.6225146
       ENTHALPY= 0.3875947E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
       DENSITY= 0.420101E-04 LB/CUFT
   VEL= 0.122E+05 FT/SEC   MACH= 0.229E+01    AREA= 0.195E+01 SQFT/LB/SEC

   SPECIES   MOLE FR.        SPECIES   MOLE FR.       SPECIES   MOLE FR.
        C  0.10802E+00             O  0.55407E+00            O2 0.12821E-04
        CO 0.33790E+00           CO2 0.30772E-05

STATION NO  20

        CP-FROZEN    CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
        0.33432E+00 0.46302E+01-0.17733E+01 0.73528E-01 0.11029E+01

      TEMP= 5543.6972 DEG-K       PRES=  0.0124 ATM      MOL WT= 19.7805939
       ENTHALPY= 0.3740616E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
       DENSITY= 0.337883E-04 LB/CUFT
   VEL= 0.127E+05 FT/SEC   MACH= 0.241E+01    AREA= 0.233E+01 SQFT/LB/SEC

   SPECIES   MOLE FR.        SPECIES   MOLE FR.       SPECIES   MOLE FR.
        C  0.10083E+00             O  0.55047E+00            O2 0.11689E-04
        CO 0.34868E+00           CO2 0.29175E-05


STATION NO  21

        CP-FROZEN    CP-EQUIL    DLNM/DLNT    DLNM/DLNP     GAMMA
        0.32729E+00 0.31623E+01-0.11129E+01 0.41443E-01 0.11019E+01

      TEMP= 4960.3313 DEG-K       PRES=  0.0020 ATM      MOL WT= 20.9454479
       ENTHALPY= 0.2802094E+04 CAL/GM     ENTROPY= 0.35512E+01 CAL/GM-DEG K
       DENSITY= 0.640370E-05 LB/CUFT
   VEL= 0.157E+05 FT/SEC   MACH= 0.324E+01    AREA= 0.996E+01 SQFT/LB/SEC

   SPECIES   MOLE FR.        SPECIES   MOLE FR.       SPECIES   MOLE FR.
        C  0.47871E-01             O  0.52401E+00            O2 0.63231E-05
        CO 0.42811E+00           CO2 0.20612E-05
```

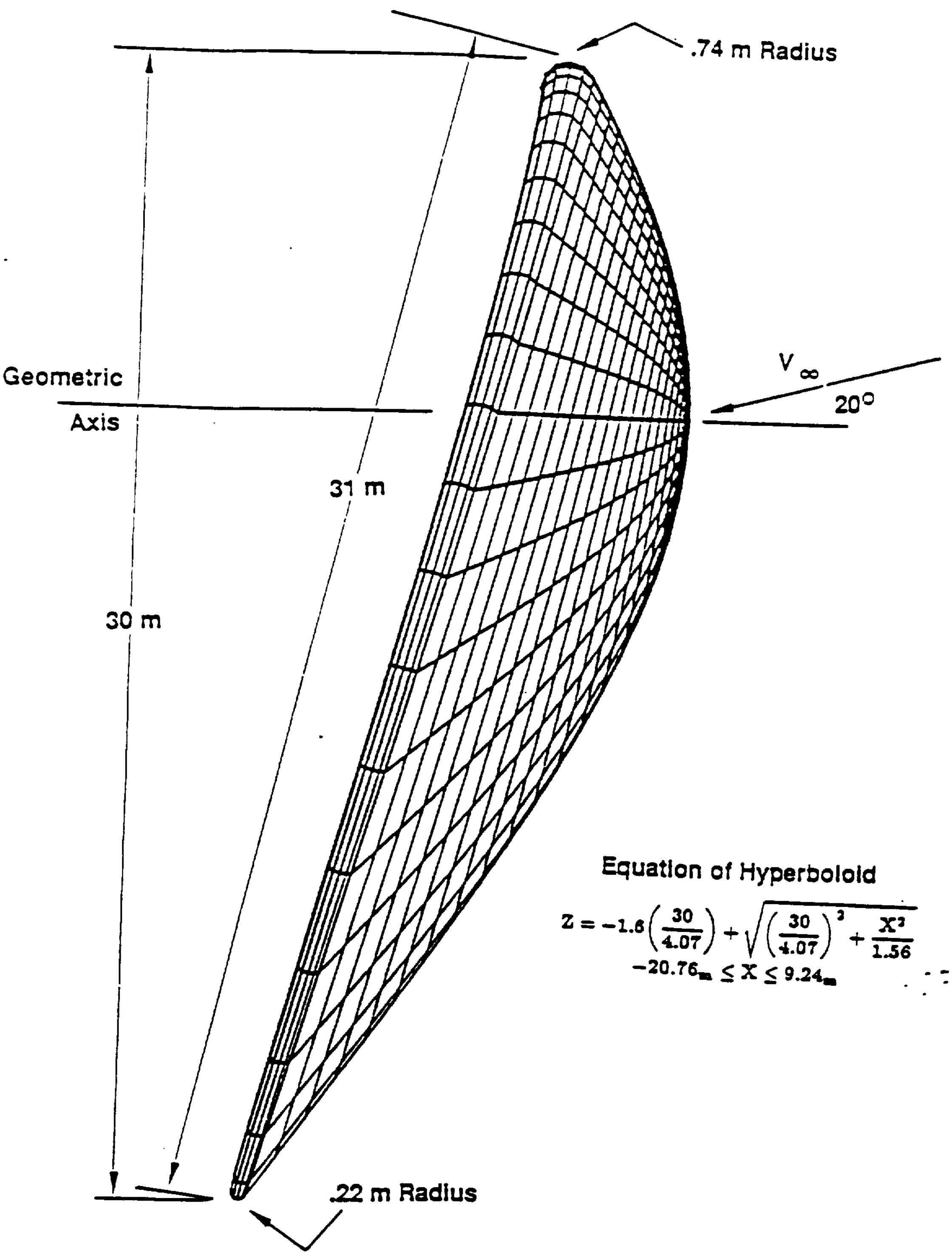


Figure 1: Side View of MTV Aerobrake Structure

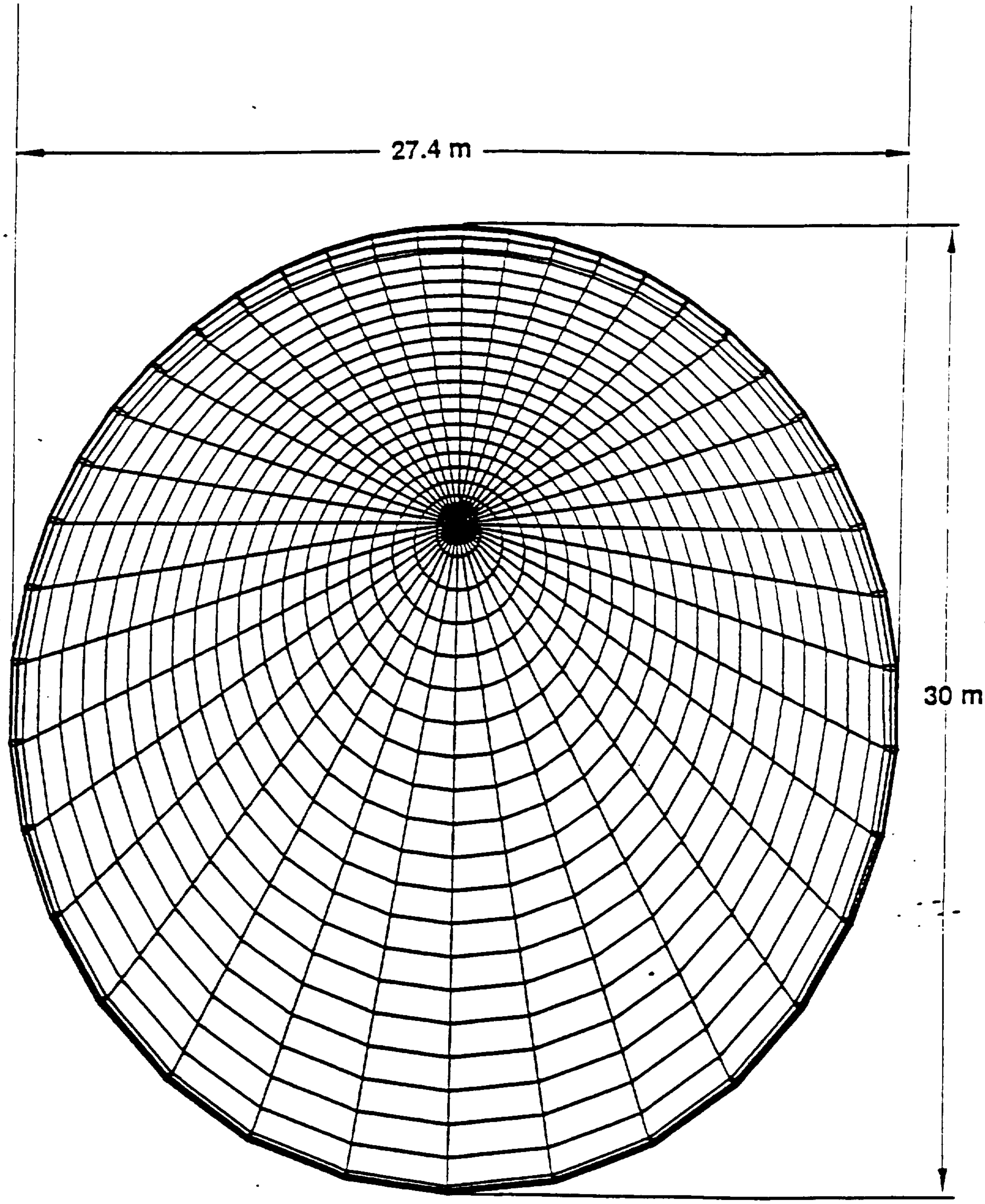


Figure 2: Top View of MTV Aerobrake Structure

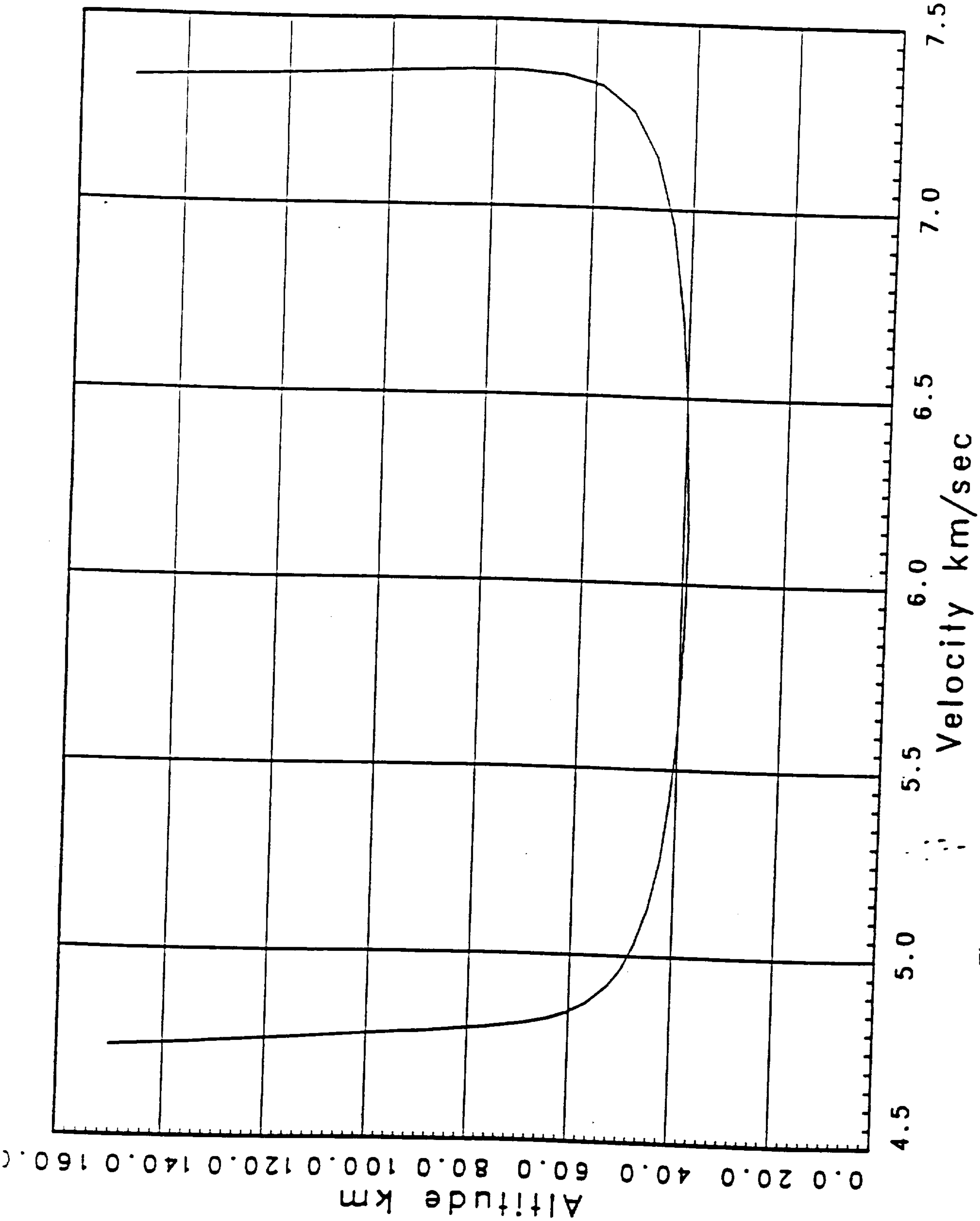


Figure 3: MTV Aerocapture Trajectory in Mars Atmosphere

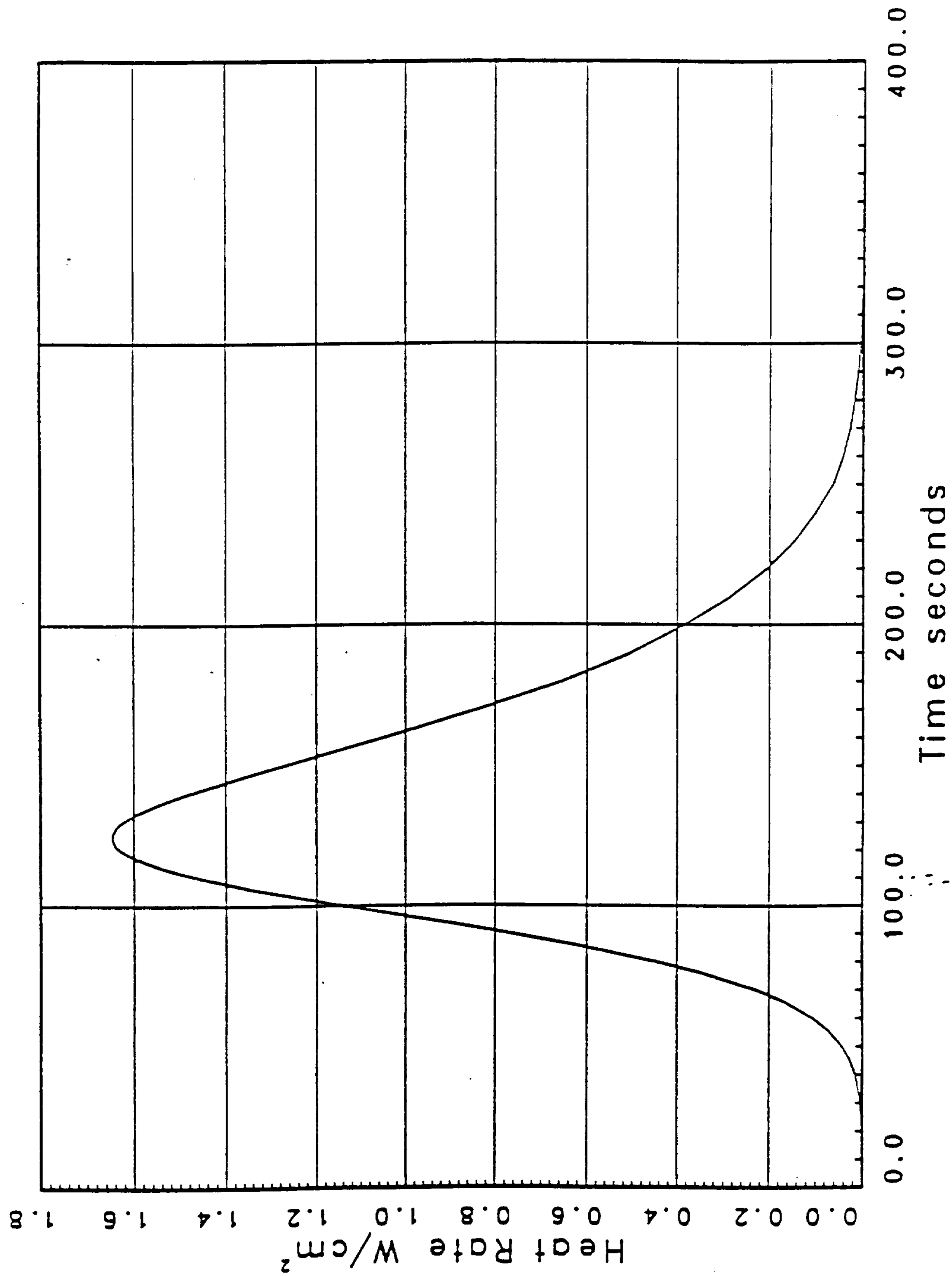


Figure 4: MTV Base Convective Heating Environment

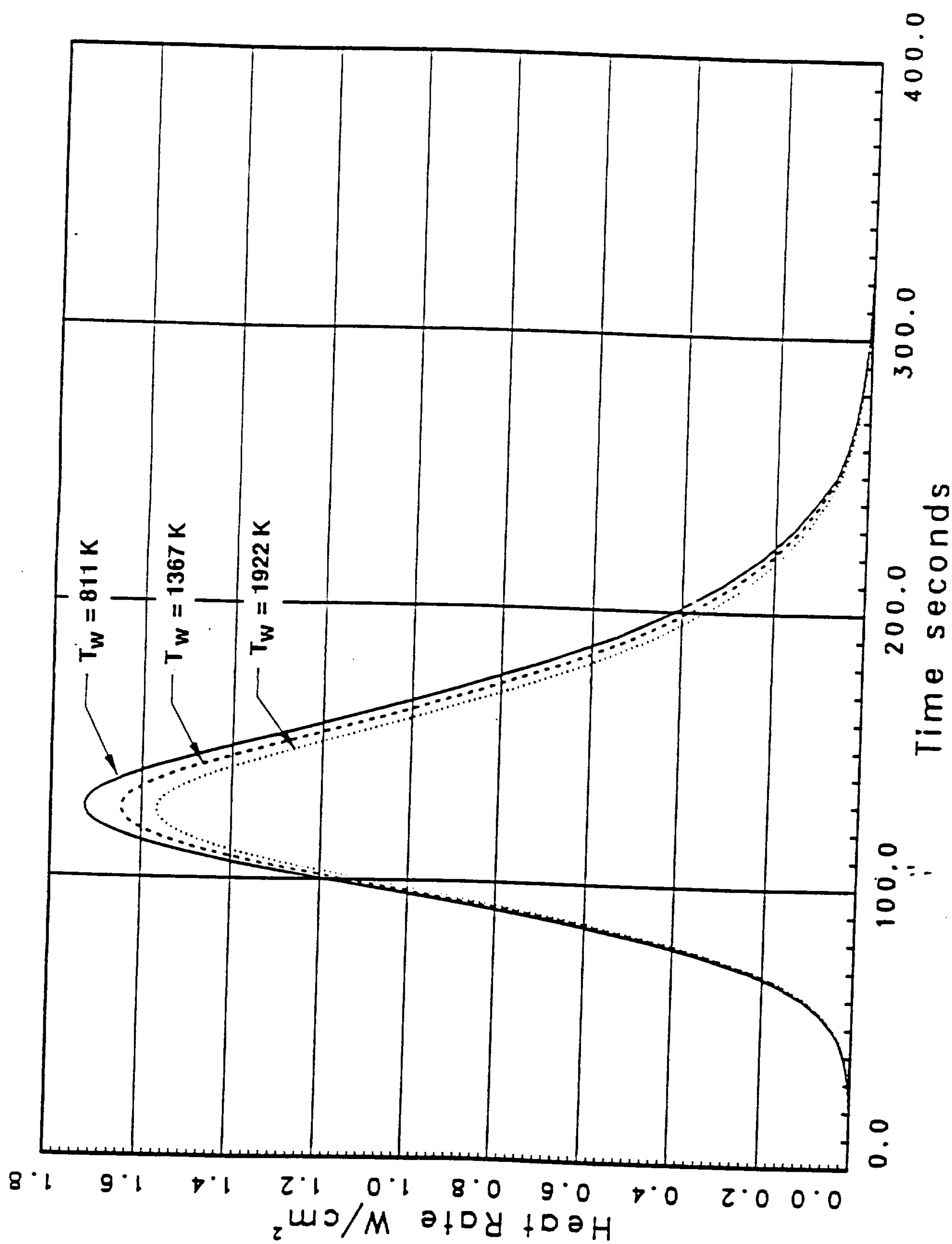


Figure 5: Sensitivity of Base Convective Heating to Wall Temperature

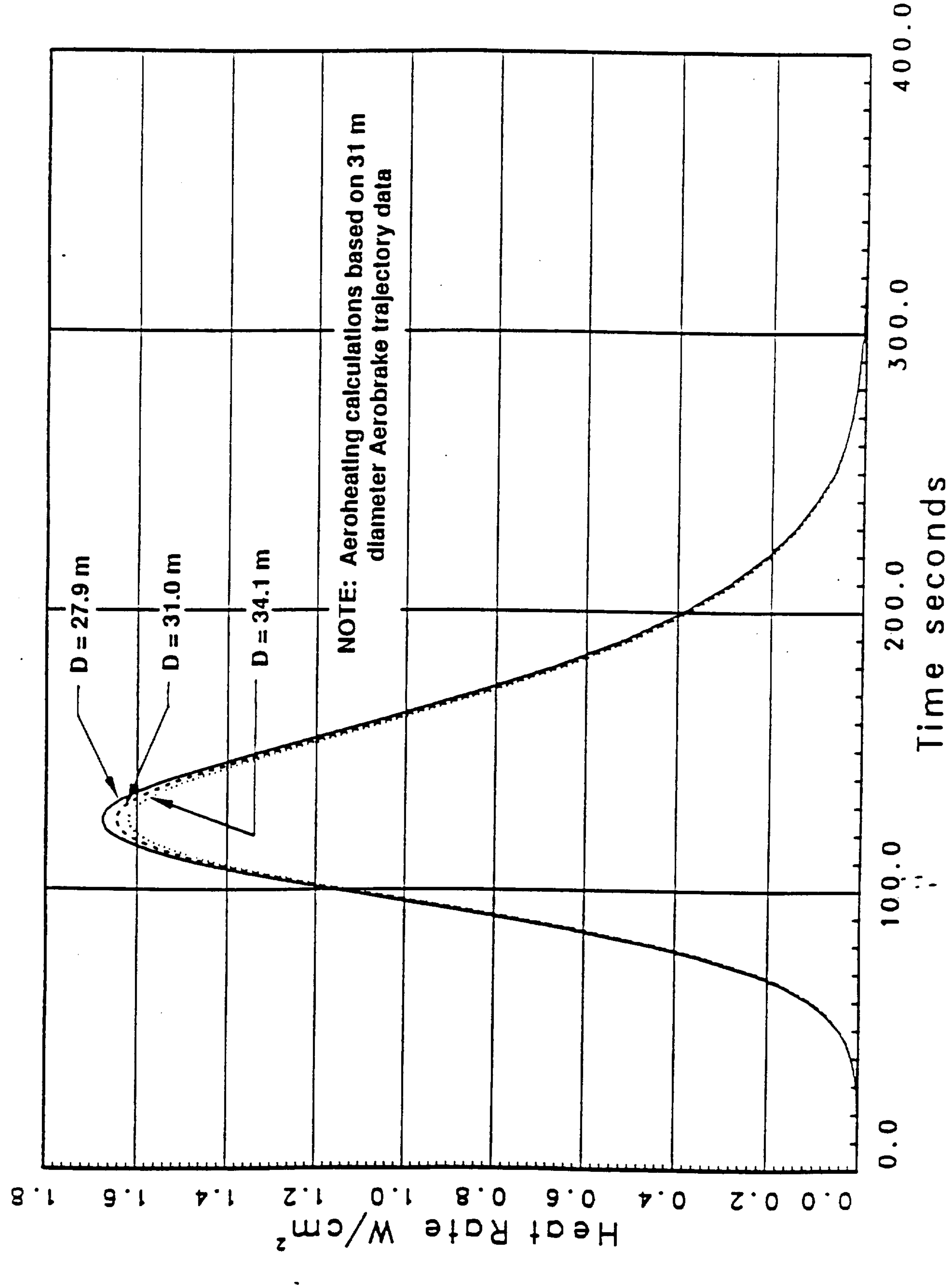


Figure 6: Sensitivity of Base Convective Heating to Base Diameter

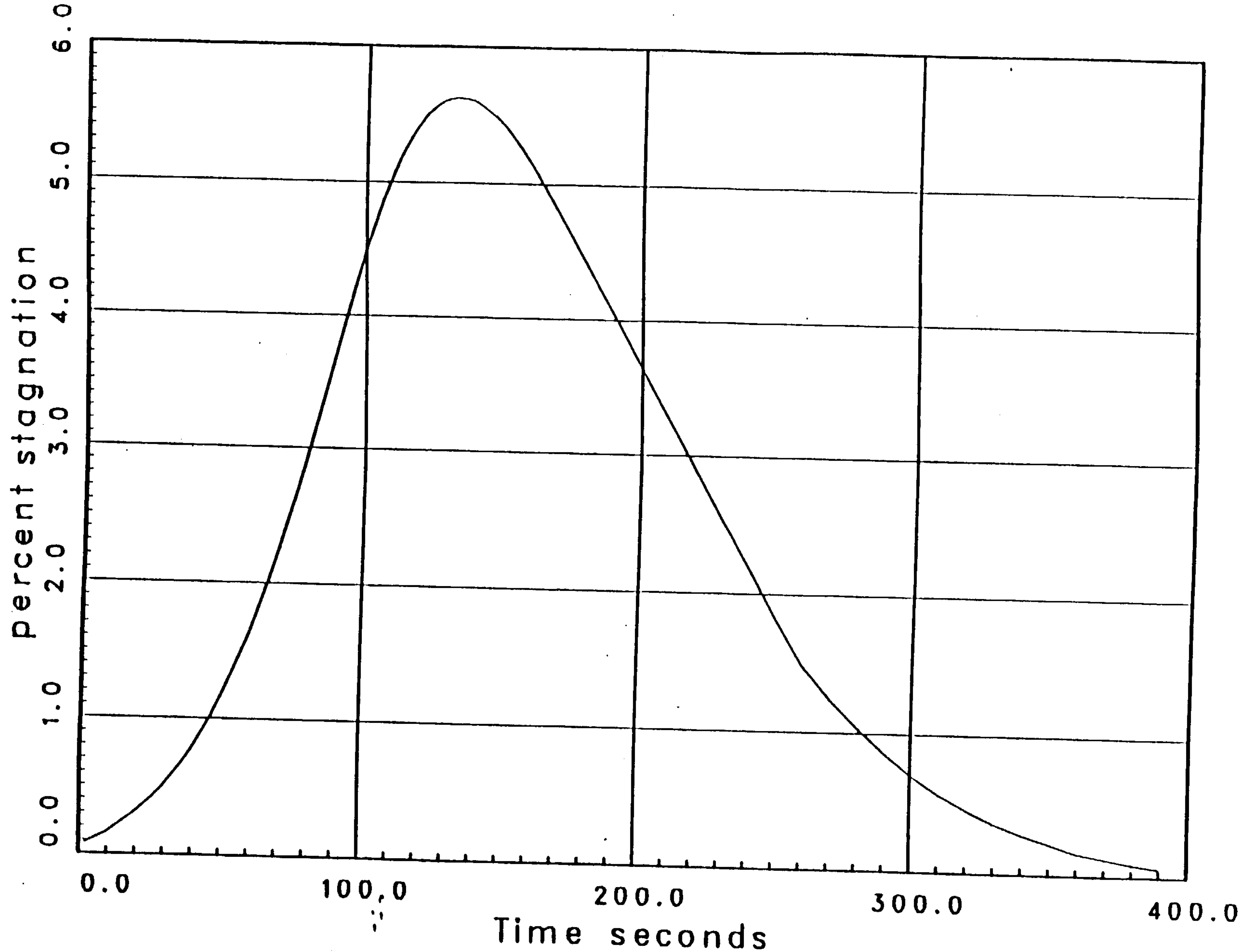


Figure 7: Base Convective Heating Environment as Percent of Stagnation Point Convective Heating

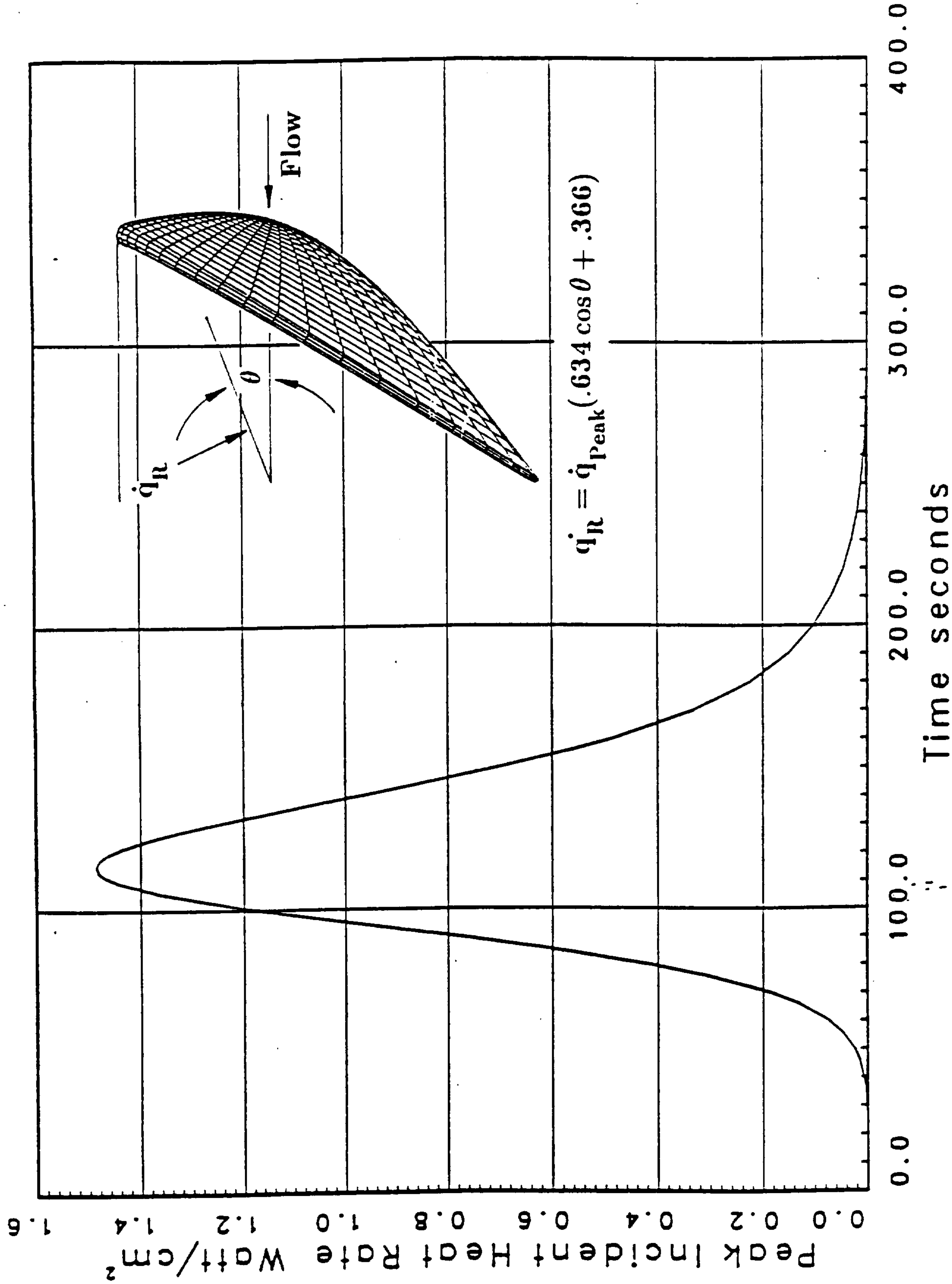


Figure 8: MTV Base Radiative Heating Environment

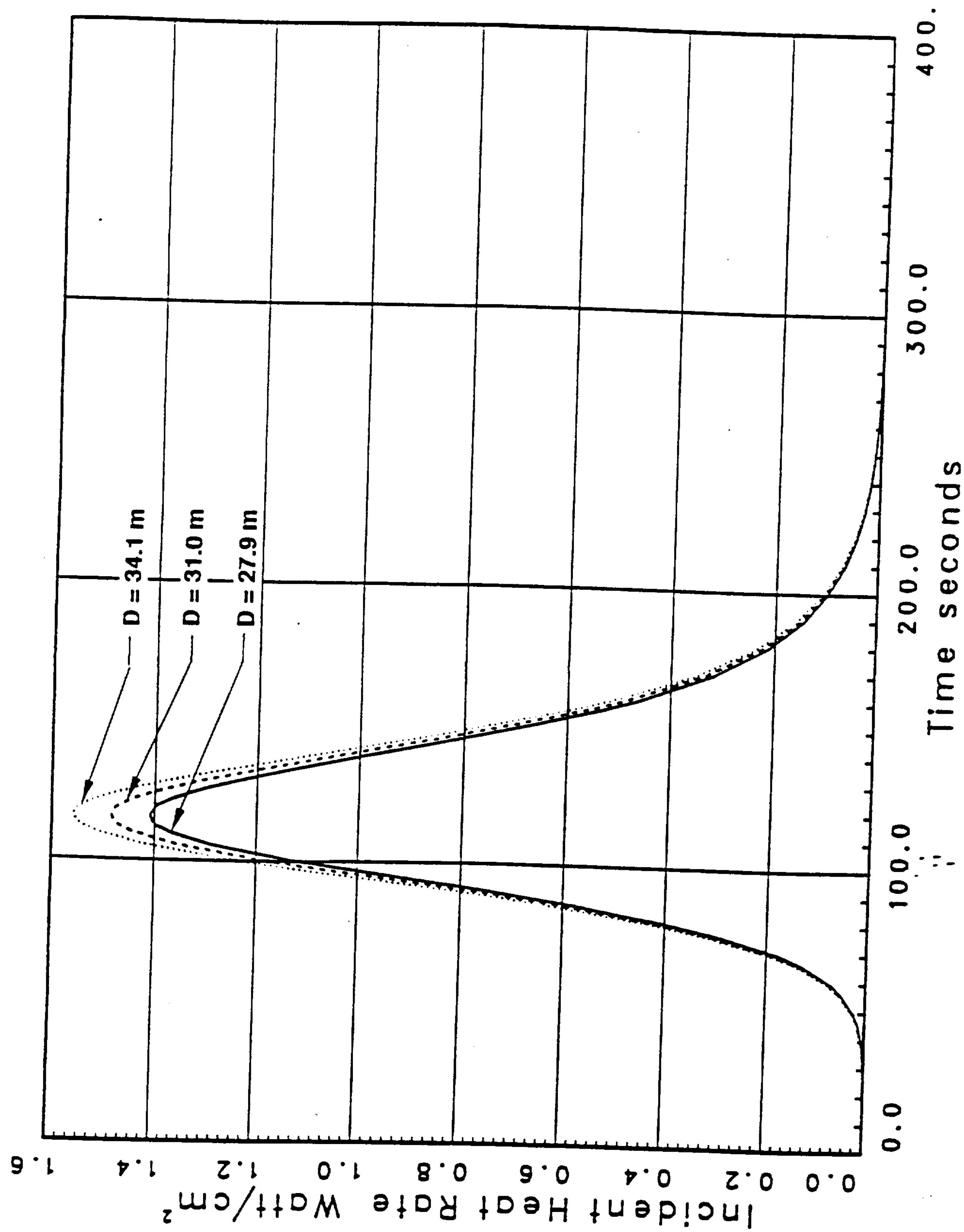


Figure 9: Sensitivity of Base Radiation Heating Environment to Base Diameter

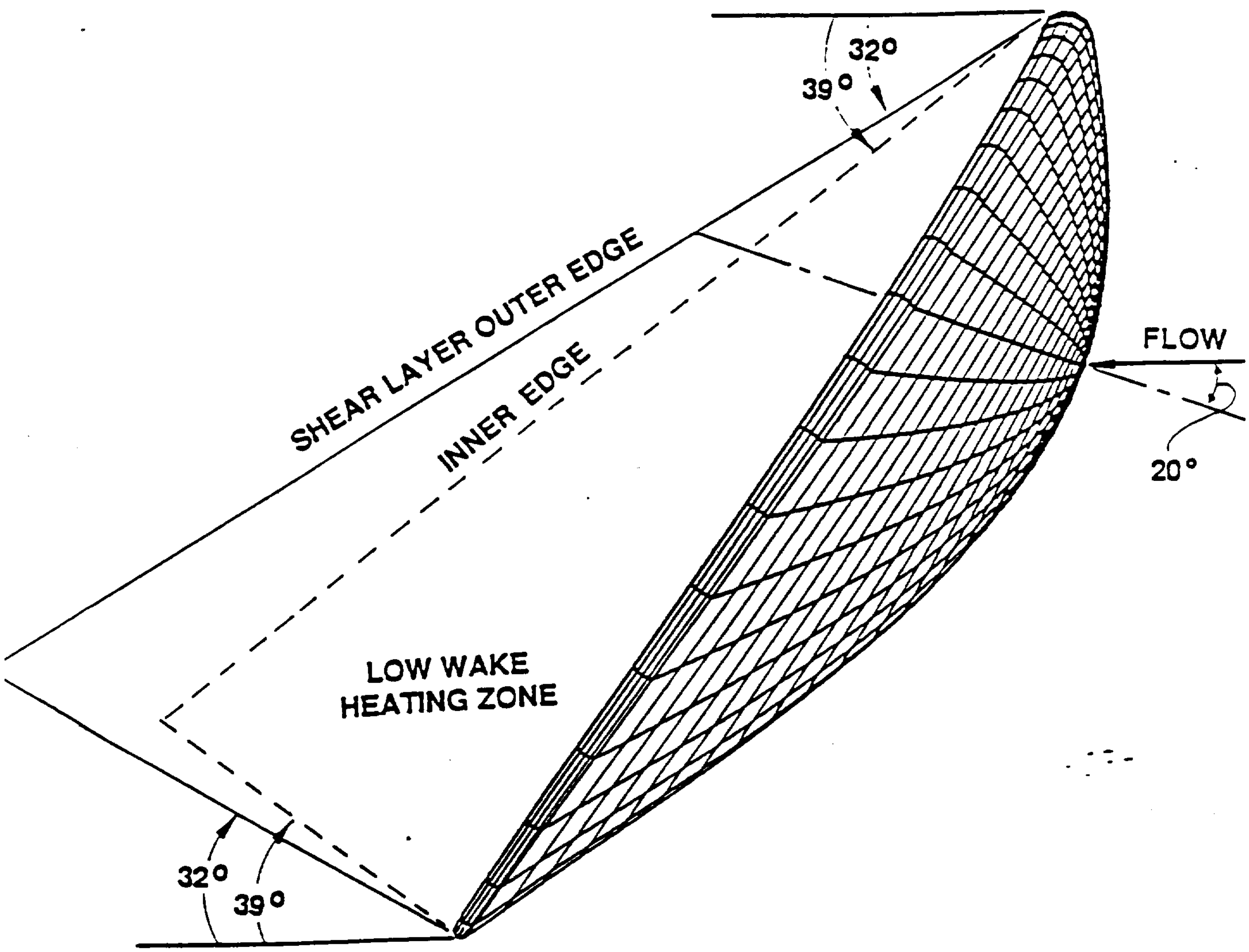


Figure 10: Wake Closure Zone at Peak Heating

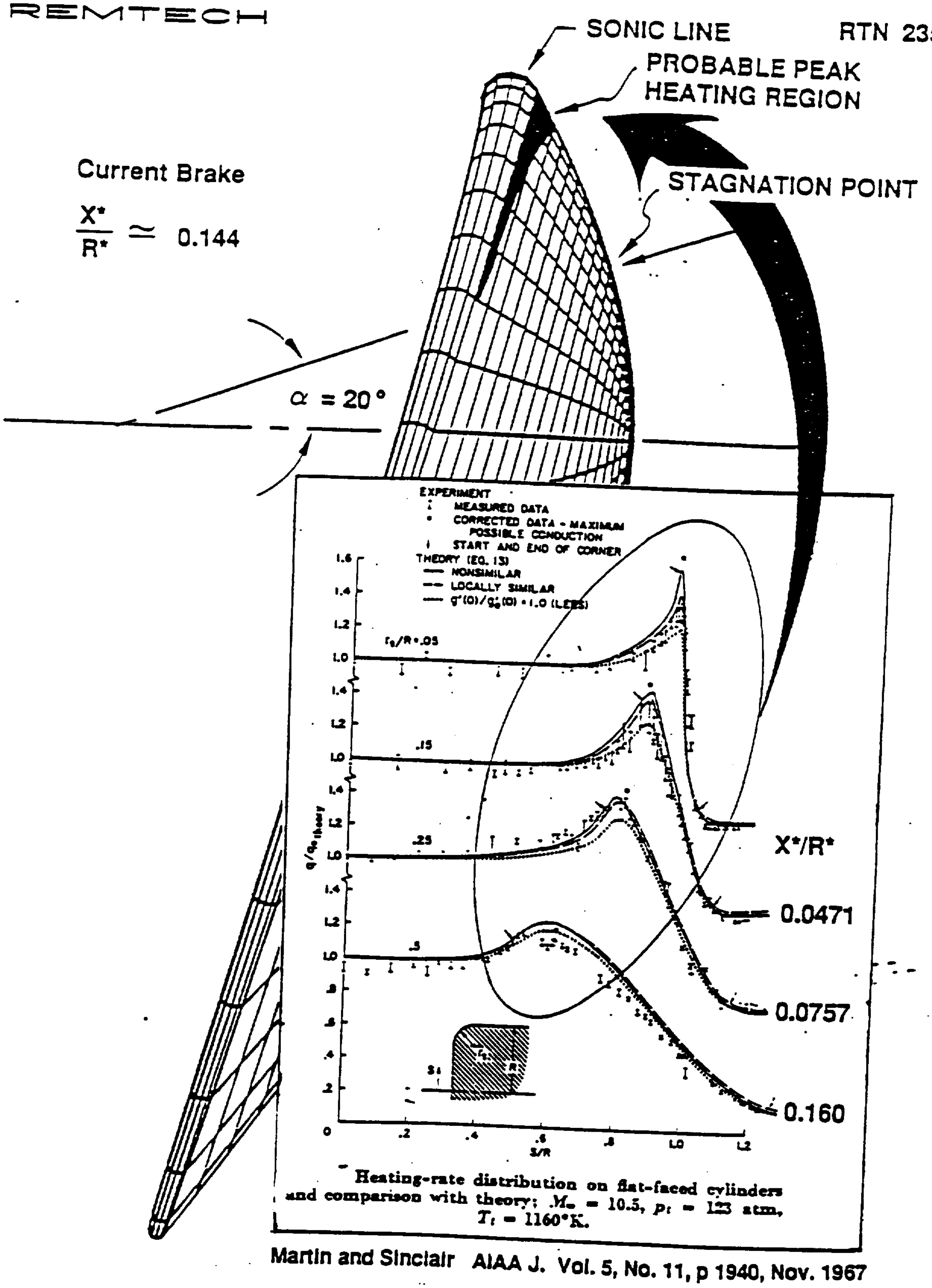


Figure 11: Constrained Sonic Point Forebody Heating

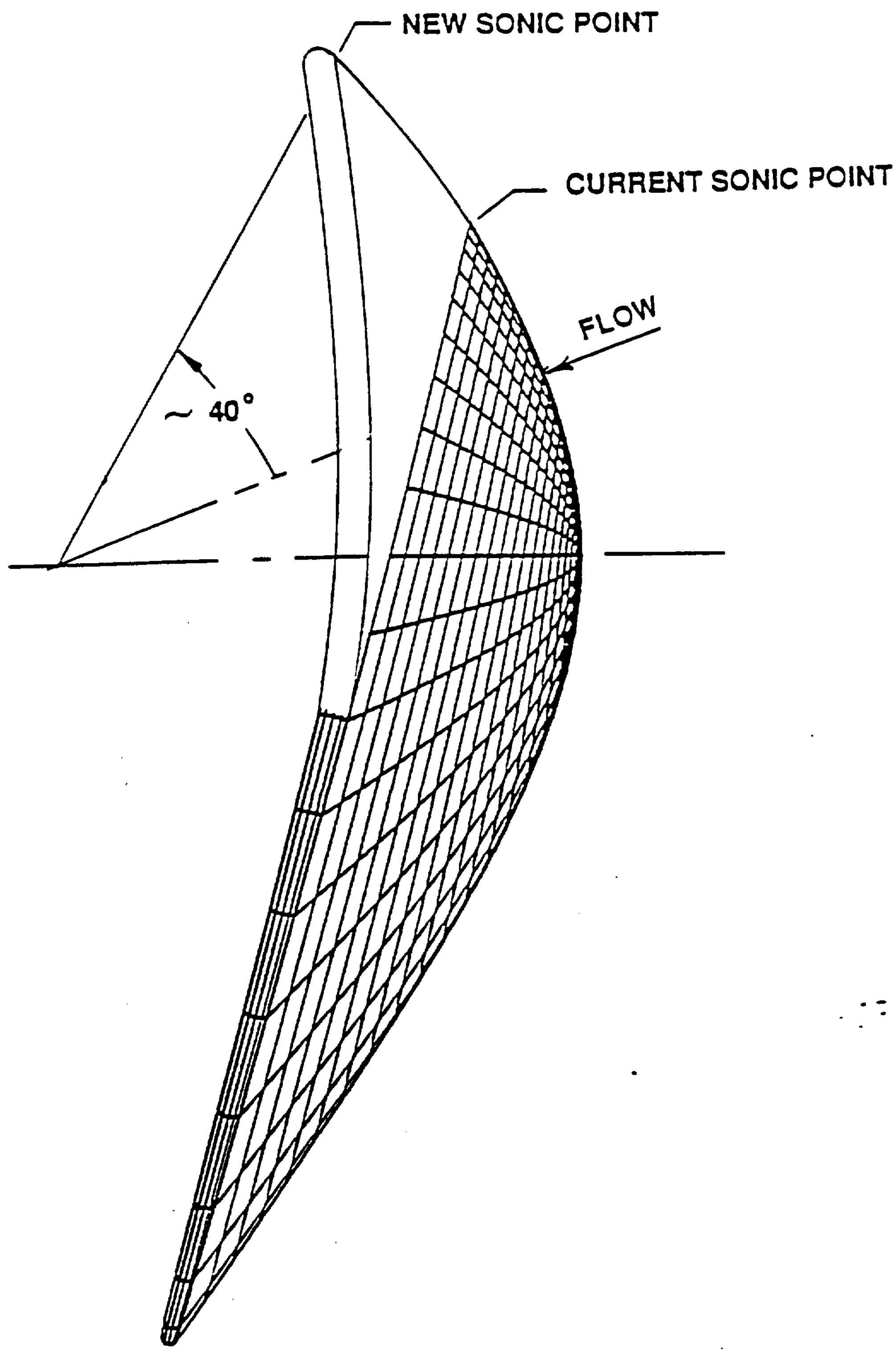


Figure 12: Suggested Brake Modifications to Reduce Peak Brake Heating and Enlarge Wake Payload Zone

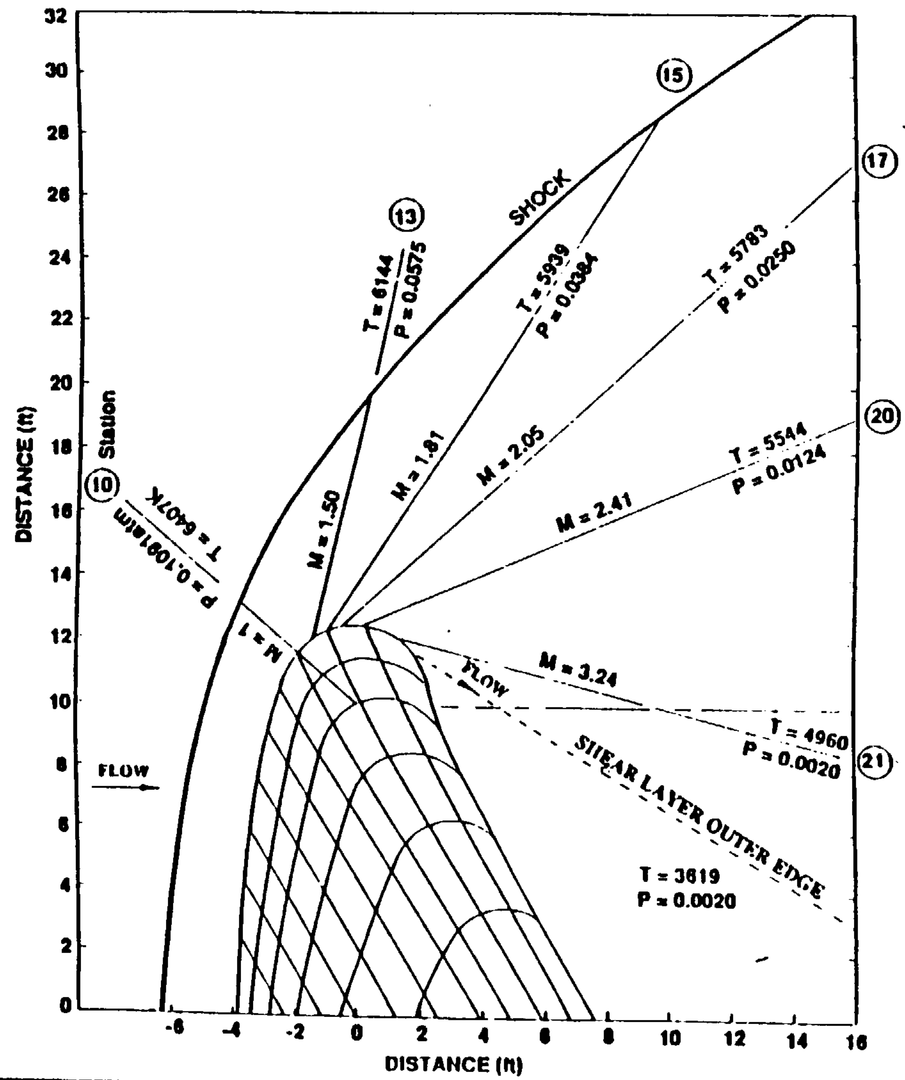
Estimated Brake Shoulder Flow Field Properties

This chart shows the expansion fan around the shoulder of the MTV. The results are from BLIMPK runs and approximate shock shapes based on AFE Schlieren photographs

— continued —

ESTIMATED BRAKE SHOULDER FLOW FIELD PROPERTIES

REMTECH



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Equipment Life and Self-Check

This section discusses the work accomplished and the issues identified concerning the processing, integration , test, assembly, verification and operation of the Mars vehicle (primarily the reference Cryo/Aerobrake vehicle). Self-check capabilities include Built-in Test/ Built -in Test Equipment (BIT/BITE), Fault Detection and Differentiation (FD&D), and monitoring techniques. The philosophy of readiness for each stage in piece of equipment's lifetime is based on these capabilities and techniques, especially for the test, verification and operations phases.

Since autonomous checking involves both inter- and intra-system "communications", interface identification and verification need to establish proper functioning. Ground processing is based on sequencing of hardware as constrained by integration, test and assembly. Assembly involves not only the manifesting and launch of the vehicle, but also the construction and interfaces of the Assembly Node. Assembly in space requires that the self- check capability be part of the process of construction and operations, since the demand in manpower otherwise needed to perform test and checkout would overwhelm any support function or the limited crew .

Test, verification and operation of the Mars vehicle puts this self-check philosophy into practice. The overall test approach utilizes self-checking at the component and system level to perform both initial and continuous testing. This imbedded self-check capability is also used in the verification phase for rechecking components and systems within the integrated elements. Operations will use continuous on-board autonomous checking during each stage of the mission. Without this capability, monitoring and statusing over the life of the vehicle would require more EVA work than the schedule can afford. This requires that the Data Management System of the vehicle (independent flight) and the Earth/ SSF based assembly control point software has the capability to interpret the BIT/BITE results, separating sensor failure from component failure, and referencing affected areas.

Self- checking will reduce the time and increase the efficiency of the testing, verification and operation phases of the Mars vehicle program. In order to best realize these benefits, the processing, integration and assembly phases must be planned in accordance.

(4)

Verification of Operational Readiness

Operational readiness will be determined by the review of data gathered from the assembly operations and the health monitoring systems aboard the vehicle and available on-orbit and on Earth. These systems will be heavily reliant on the development of a self-check capability. This will require the development not only of the sensor system, neural network software and reliable computing capability , but the procedural methods to use this technique with confidence.

Simple functional tests will be performed on major items (such as thrusters , reactors, electrical power systems other than life support, etc.) that must be verified operational immediately prior to TMI commit.

The life support system of the MTV transit habitat will be tested-in-use as the pressurized volume will be used as an on-orbit command center during the assembly phase.

Verification of Operational Readiness

- Verification of readiness will be done by system self-check
  - will provide a computer record of procedure and responses
  - will reduce the man power required to test
  - will provide a rapid recheck capability
  - system will differentiate between a system failure or a sensor failure
  - verification will be done from several on-board command stations and from Earth by telemetry

- Continuous on-board vehicle health monitoring from system online function must be provided.

- "Burp" testing of engines and thrusters at SSF safe distances

- On-board testing and evaluation of ECLSS by use of the MTV/ MEV prior to TMI

- The test philosophy will be geared to producing a verified system without redundant testing

Mars Aerobrake Assembly and Integration at the VAB

The current Vehicle Assembly Building (VAB) has a transfer aisle with the dimensions of 92.5 feet the aerobrake is 91.8 feet, as shown in the following chart, the Aerobrake can be transferred through the transfer aisle horizontally.

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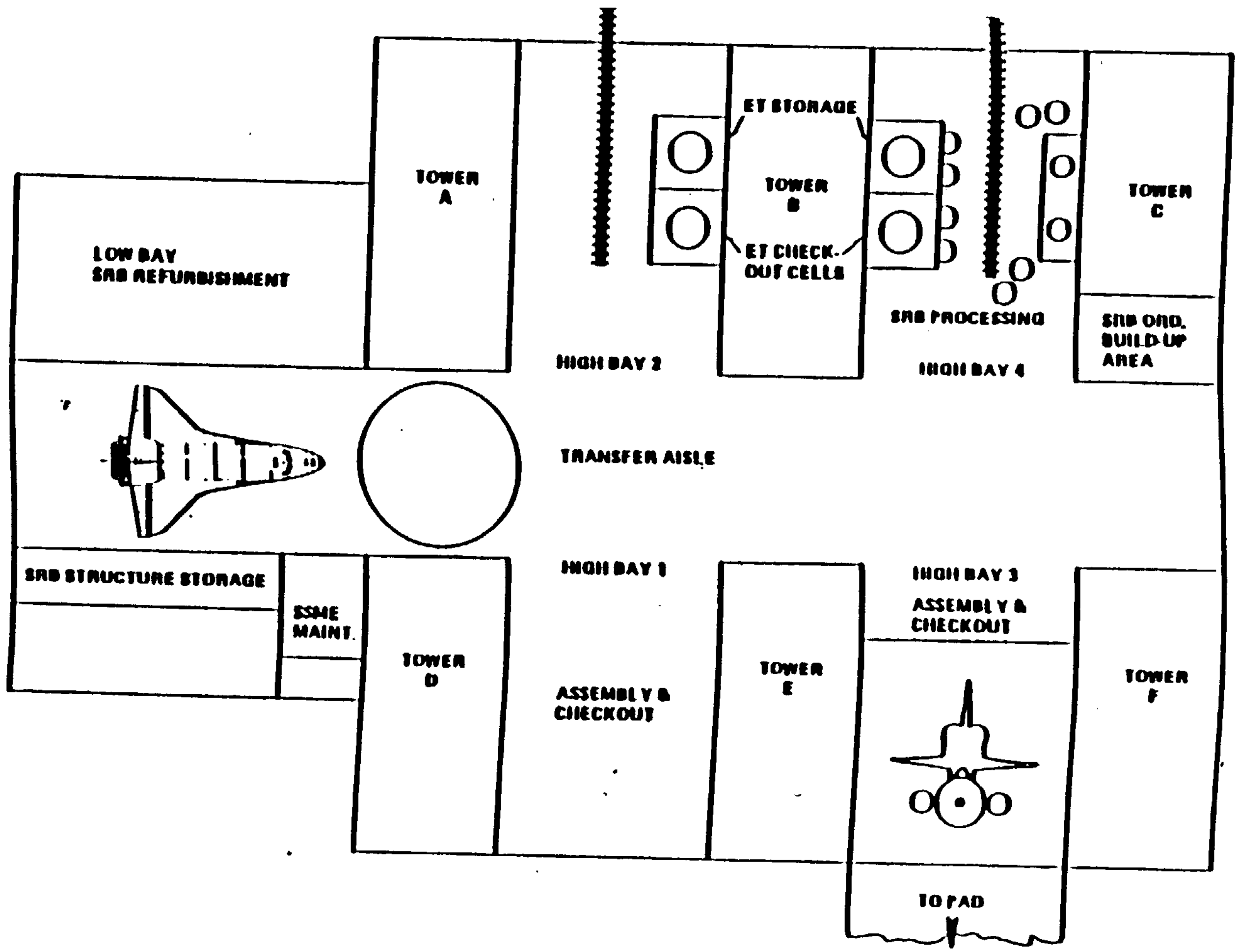

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Mars Aerobrake Launch Option

Modifications of existing equipment will be required to assemble and check out the Integrated Launch Option in the current VAB High Bay.

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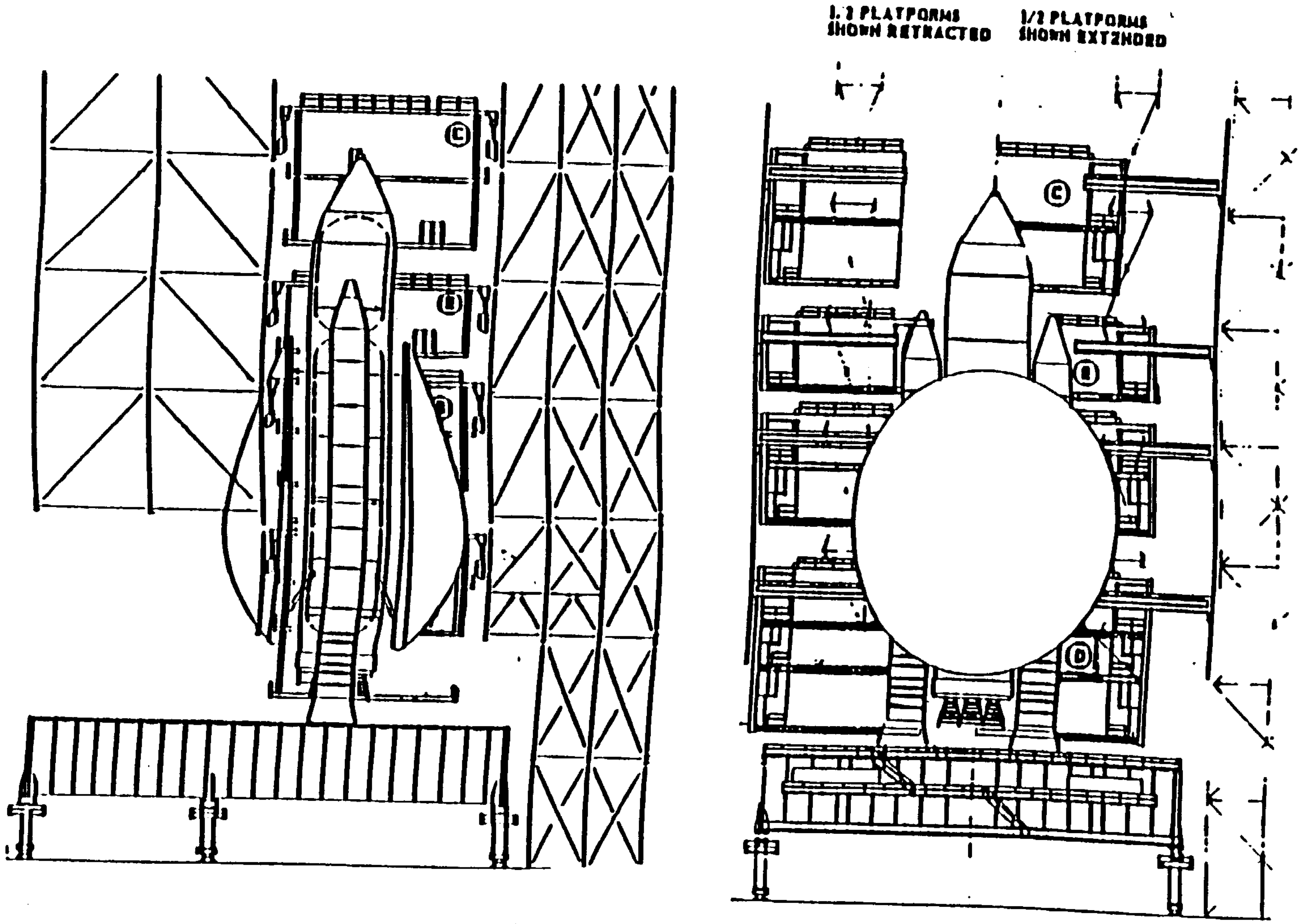
Aerobrake Launch Option



VAB High Bay

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Launch Site Impacts

The following Launch Site Impacts were derived from current facility and equipment limitations. Refurbishment and modification of existing equipment or construction of new facilities will accommodate an Integrated Launch Option.

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[illegible] / Integral Aerobrake
Launch Site Impacts

- Transporter required for fully assembled Aerobrake
- VAB High Bay access platforms will require modifications
- Aerobrake will prohibit use of the Rotating Service Structure without major modifications
- Fixed Service Structure swing arm extension and retraction may interfer with the Aerobrake
- Large Aerobrake cross-sectional area will impart large wind loads to the launch vehicle
  - Increased loads to hold down fixtures
  - Revised launch commit criteria for maximum winds at launch

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Cryo/Aerobrake Mission Operations Outline

This is a top level outline of the major task sequences and their relative location for the Cryogenic fuel/Aerobrake vehicle for Mars missions. It is divided into four segments:

a) Near-Earth Operations (initial)- involving operations from hardware buildup to Trans -Mars Burn

b) Transit Operations - operations to be performed during the outbound transit flight from Trans-Mars Burn through the MTV/MEV separation for Mars atmosphere aerocapture

c) Mars Operations - covers the events from MEV and MTV aerocapture through the Trans -Earth Burn

d)Transit and Near Earth Operations - looks at the inbound transit return to Earth and capture operations at Earth

These segments will be further broken down into distinct top-level tasks in the Mars Operations Task Flow for the Cryo/Aerobrake.

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Mars Major Mission Operations
Cryo-Aerobrake

Near- Earth Operations:

- Ground testing and support
- Launch support and Launch
- On orbit assembly and checkout
- Departure positioning and TMI burn

Transit Operations:

- TMI stage drop
- Transit flight configuration
- Optional maneuvers- Venus swingby, Deep Space Burn, coast correction and reconfigure
- Periodic Maintenance and inspection
- Autonomous checks and separation maneuvers

Mars Operations:

- Aerocapture, rendezvous and dock
- Separation
  - MEV: Land and establish base, ascend
  - MTV : Autonomous orbit, communications relay, survey
- Rendezvous and dock
- Jettison excess mass
- TEI burn

Transit and Near Earth:

- Transit configuration
- Optional Maneuvers- coast correction , Venus Swingby, reconfigure
- Earth Return- ECCV direct entry or ECCV orbit capture or MTV capture

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Mars Mission Operations Task Flow
Cryo/Aerobrake

This is a task by task top-level operations task flow for the Cryo/Aerobrake to complete a general Mars mission. Those tasks that are optional (may or may not be used on any given mission) are shown with darkened background. Those operations flows that present alternatives t o the baselined flow or are alternative actions are shown with dashed flow lines.

The assumptions under which these flows were developed are these: Communications with Earth is periodic, for the most part the decision making capability is with the astronauts beyond the TMI burn. This is due to the long time lag in communications (up to an hour) with Earth. Critical maneuvers such as flybys, Deep Space Burns, craft separations and docking, landing and ascent, and TEI burn will be reported to Earth prior to and after the event. Autonomous system self check capability will be present on all vehicle systems used for assembly and maintenance. Robotic assembly of the vehicle- MTV and MEV done at SSF/ on orbit, TMI assembly and integration, propellant top-off and final inspection and checkout done off-station. and the manned transfer vehicle is self-suffient in repair capability

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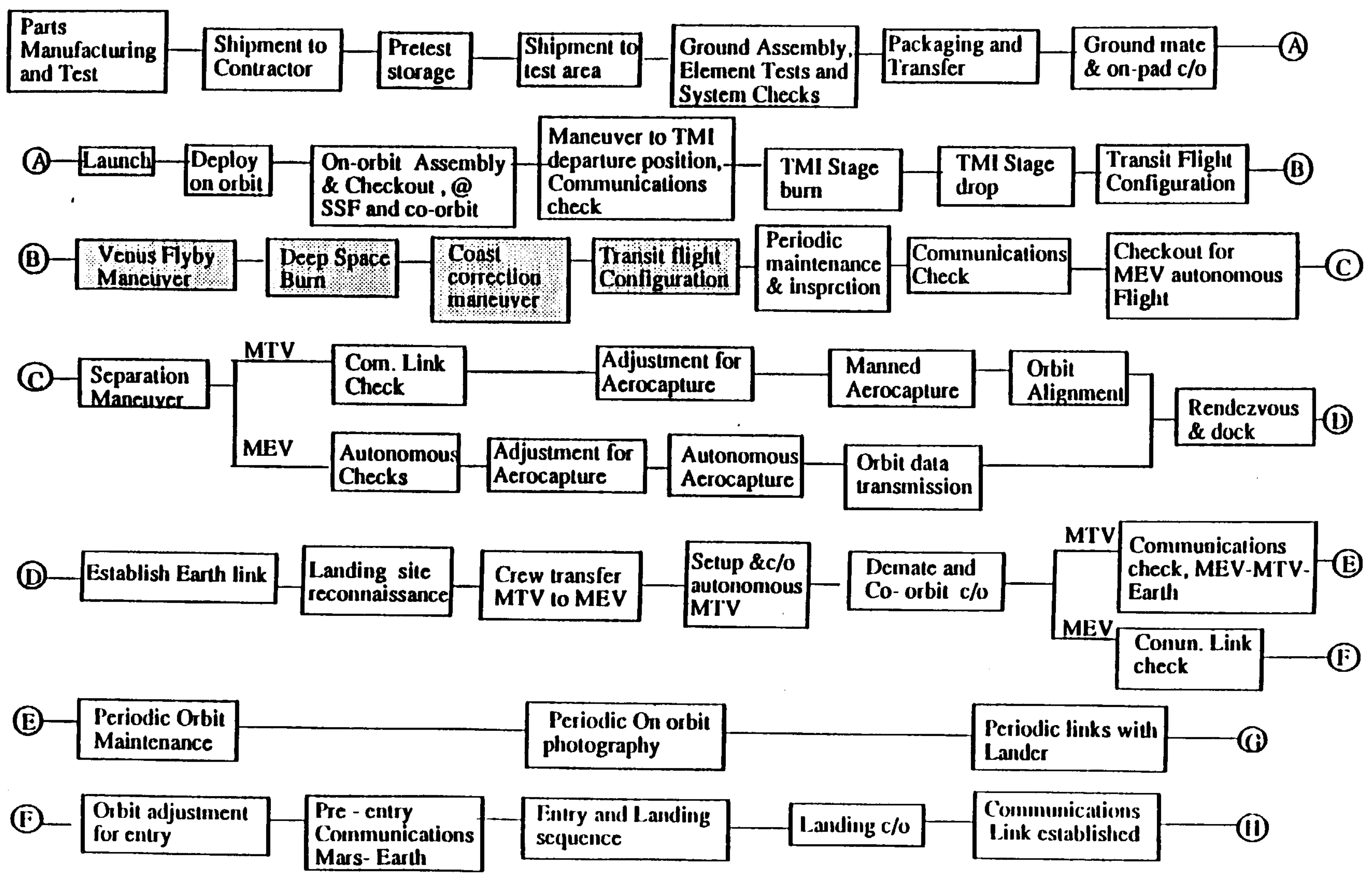
Cryo/Aerobrake



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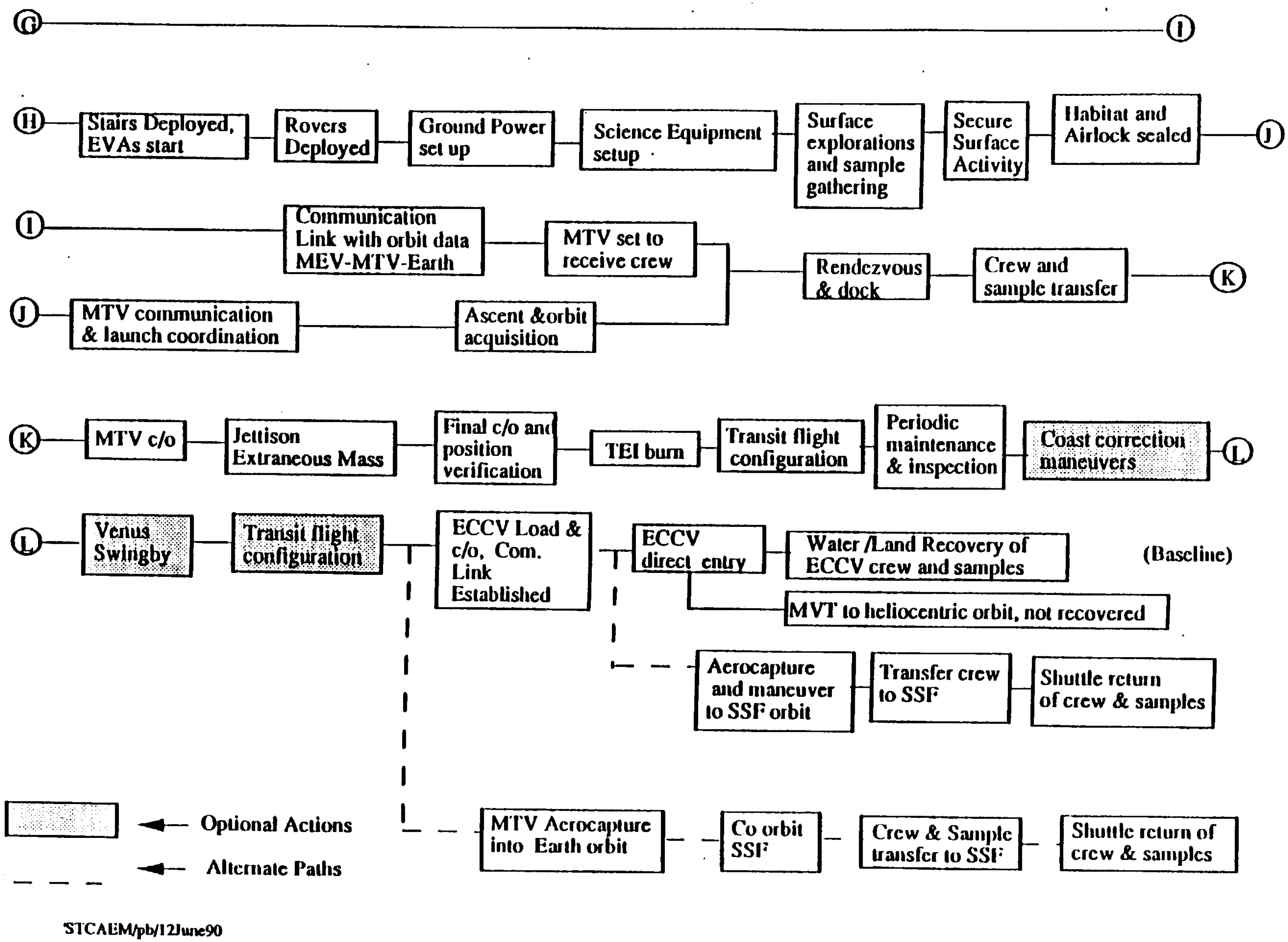

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Ground Assembly and Check Out

The following chart shows the Ground Rules and Assumptions developed for Ground Assembly Analysis of the Cryo/Aerobrake Vehicle.

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F... ...ound Assembly

- A System is a group of components and supporting structure that is integrated by a contractor and delivered as a unit to the processing facility (eg. MEV Aerobrake, MEV Descent Lander, Ascent System, etc.).
- System Interfaces are those which transmit data, power, or fluids across the system's boundaries and mechanically secure one system to another.
- Subsystems Interfaces are those which are internal to a System.
- Subsystem Interfaces are verified by the manufacturer prior to System integration.
- Component Interfaces are those which are internal to a Subsystem.
- Component Interfaces are verified by the manufacturer during Subsystem Assembly.
- Interfaces verified prior to System Level Integration will be accepted with no repetition of tests.
- Flight Hardware will be used to verify System Interfaces.
- Ground facilities will simulate assembly node operations and limitations.

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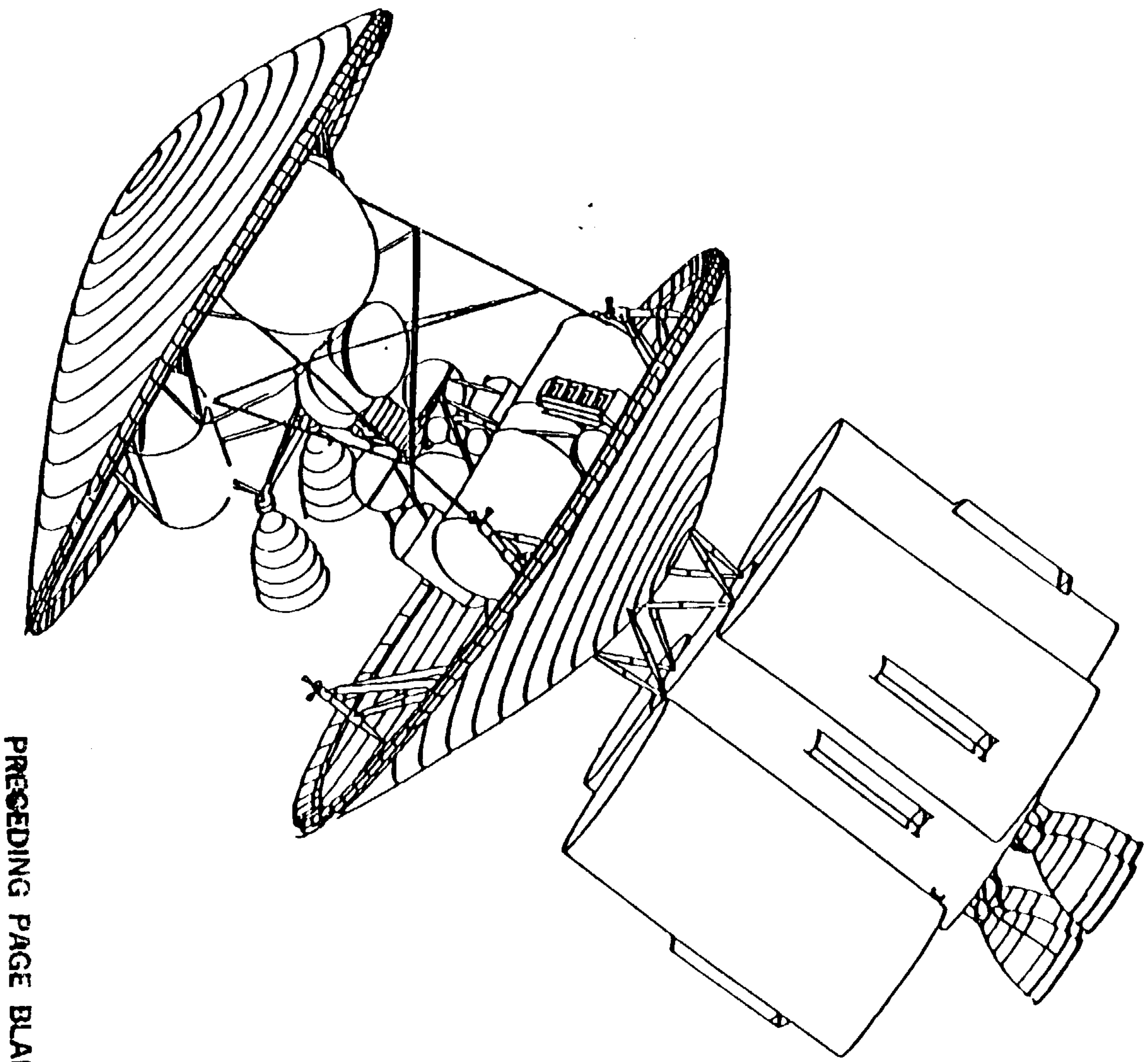
MMV Cryo/Aerobrake Vehicle



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MMV Top Level System Interfaces

The following chart shows the Top Level System Interfaces of the Cryo/Aerobrake Vehicle.

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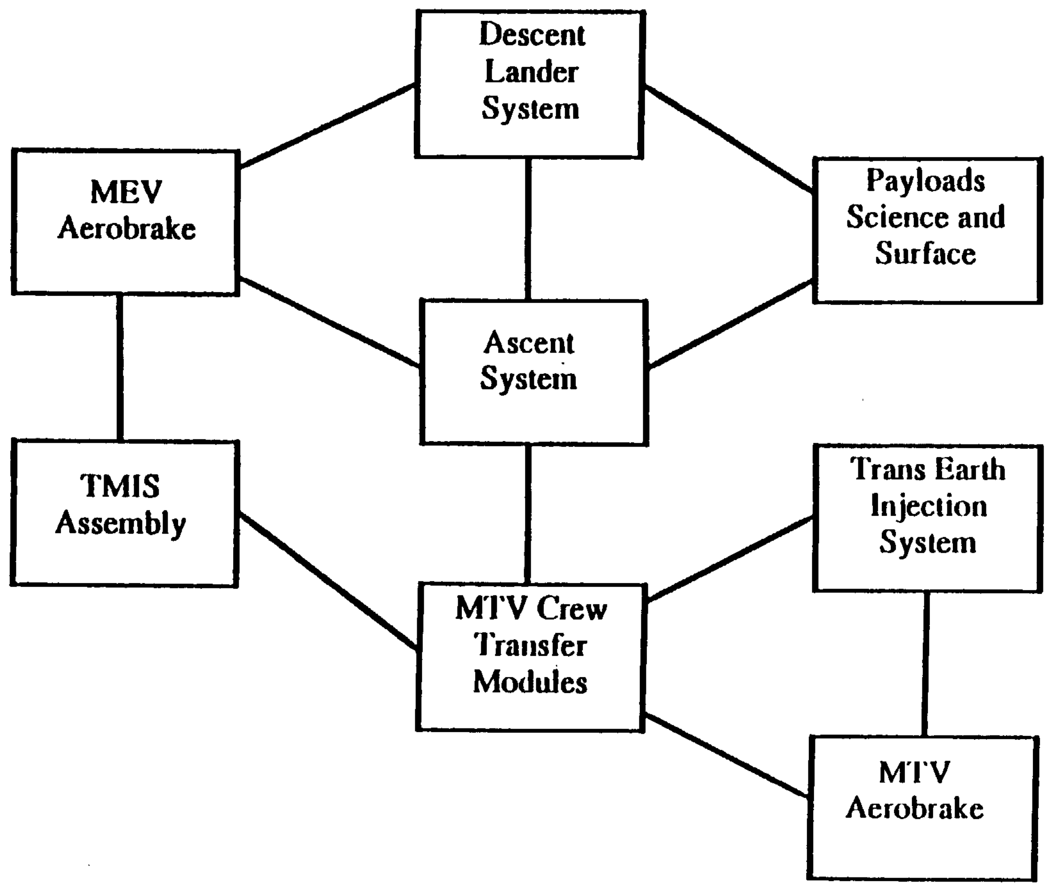
MMV System Interfaces



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MMV System Interfaces

The following two charts show Cryo/Aerobrake System and Subsystem Interfaces. The interfaces shown are major interfaces, that is, one electrical interface may represent several electrical cables. The total number of component level interfaces has not been defined at this point.

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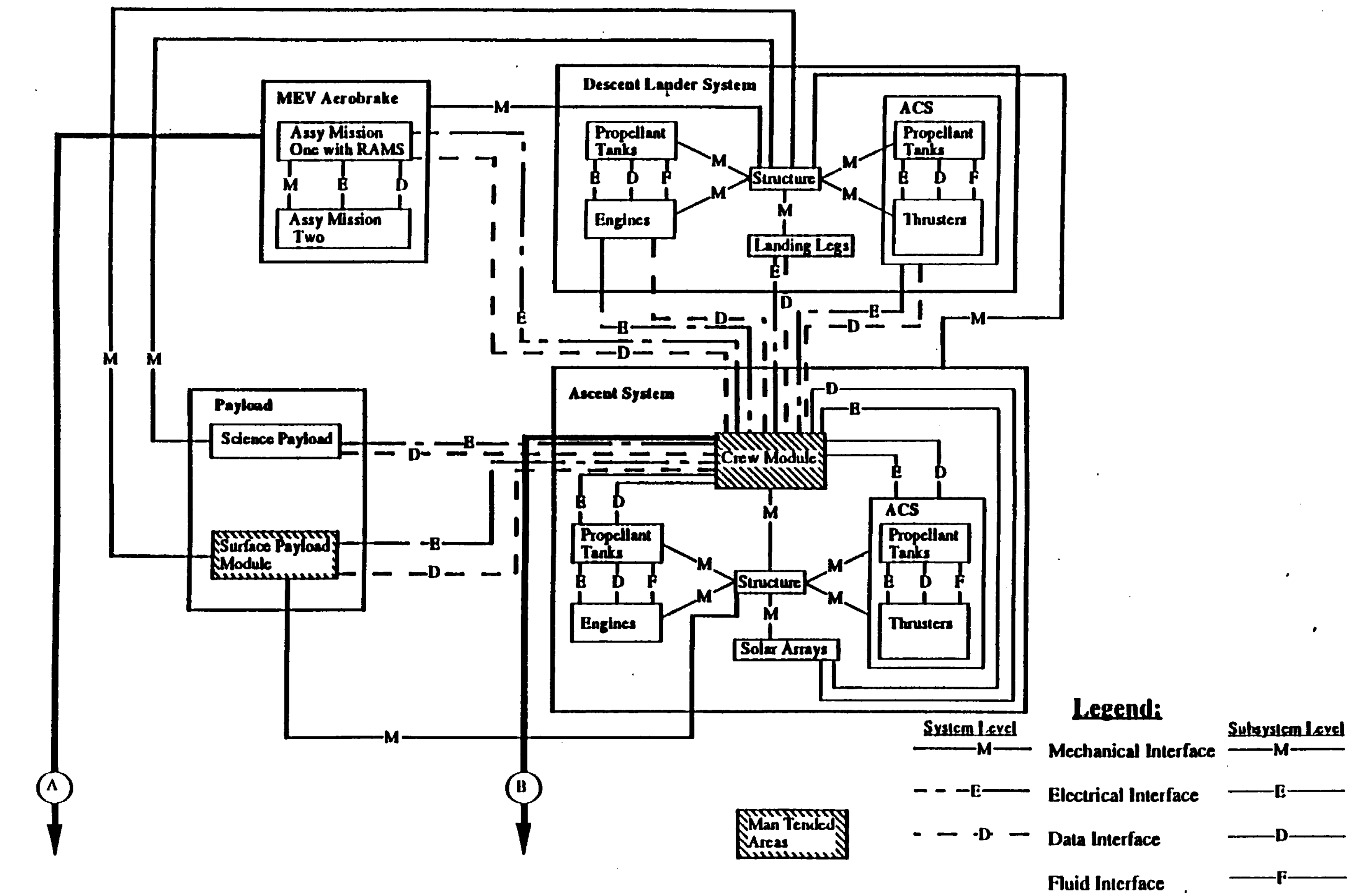
MEV System Interfaces



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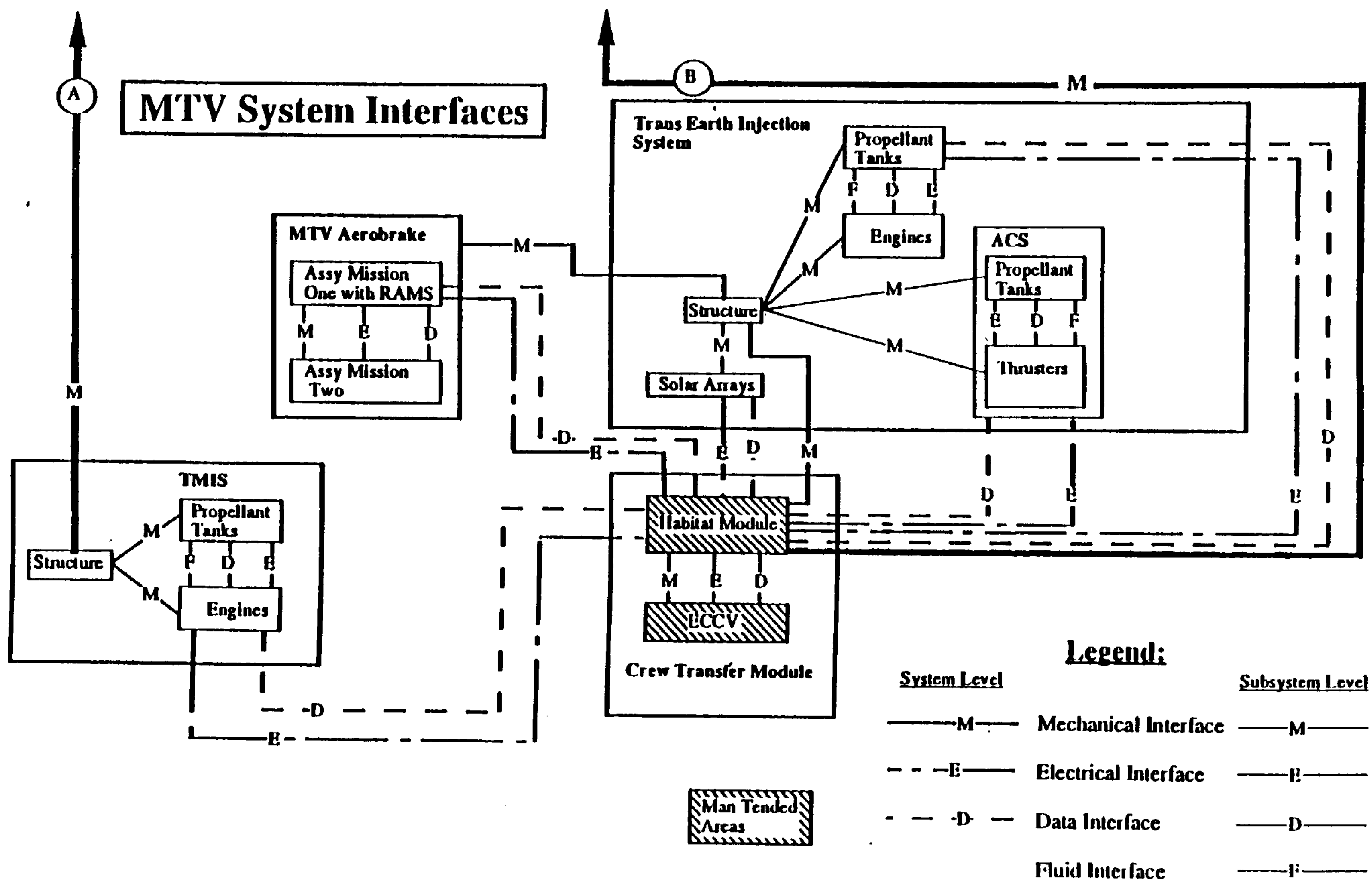

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Assembly Node Interfaces

The following chart shows the Assembly Node interfaces to Cryo/Aerobrake. The Assembly Node requirements and equipment interfaces were developed by the On-Orbit assembly analysis.

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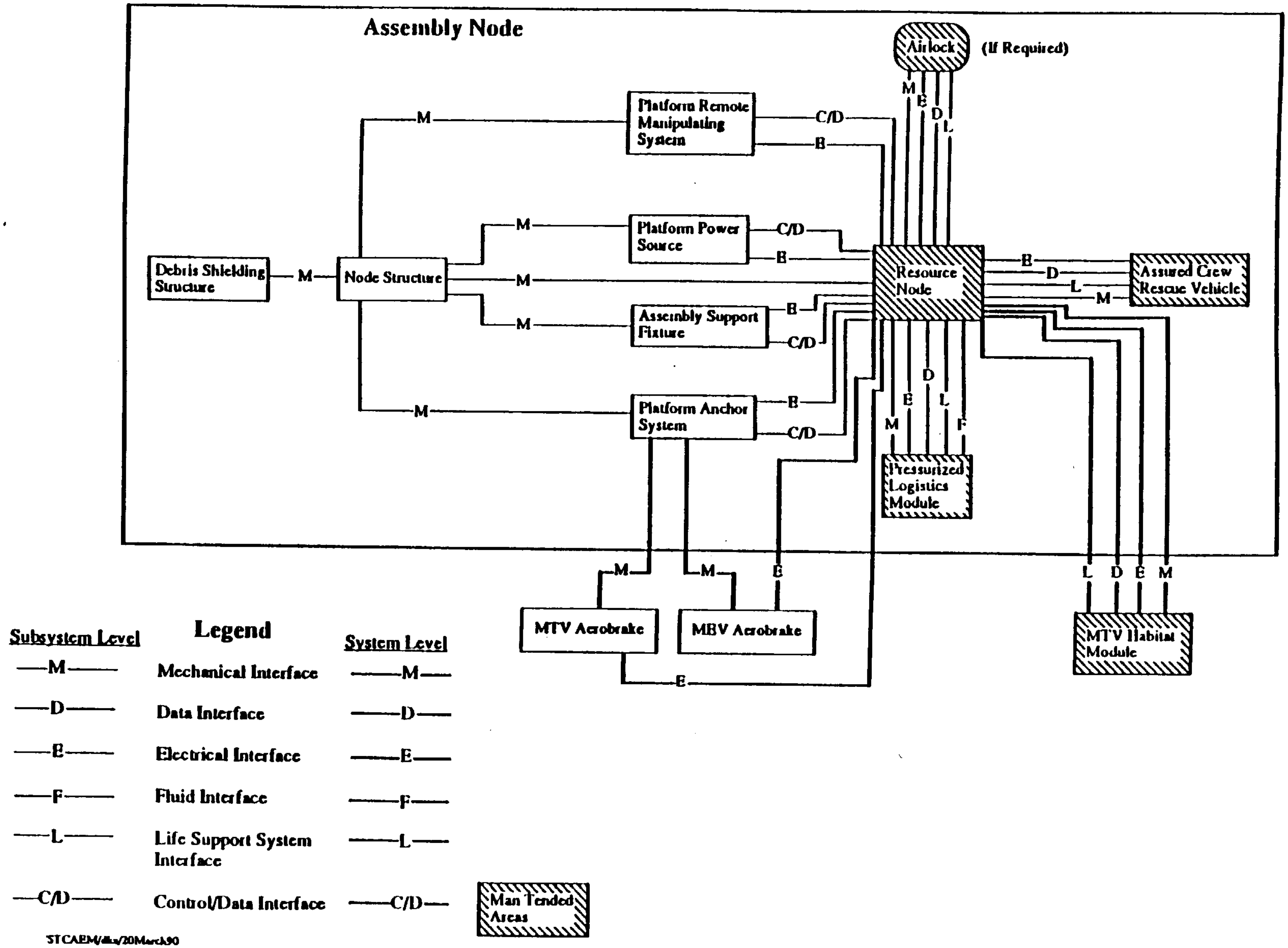
Assembly Node
Airlock
(If Required)
Platform Remote Manipulating System
Platform Power Source
Debris Shielding Structure
Node Structure
Assembly Support Fixture
Platform Anchor System
Resource Node
Assured Crew Rescue Vehicle
Pressurized Logistics Module
MTV Aerobrake
MBV Aerobrake
MTV Habitat Module
M
E
D
L
B
F
C/D
Subsystem Level
Legend
System Level
Mechanical Interface
Data Interface
Electrical Interface
Fluid Interface
Life Support System Interface
Control/Data Interface
Man Tended Areas
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Sequential Interface Verification

The following chart defines the process of Sequential Interface Verification for Ground Processing of the Cryo/Aerobrake Vehicle.

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Sequential Interface Verification

- Process of verifying the interfaces of the Mars Mission Vehicles elements without complete assembly.
- Elements are received and inspected at the assembly area.
- Internal test performed and certified by the contractor will not be repeated.
- Elements will be assembled to the level required to verify the interfaces from one element to another.
- Interfaces will be verified by flight hardware when feasible or by match mate devices/prototypes when necessary.
- Elements will be disassembled to payload configurations and processed for launch.

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Ground Processing Functional Flow

The following three charts show the Functional Flow of Ground Processing for the Cryo/Aerobrake Vehicle. This Flow is a top level flow that shows the requirements for sequential interface verification.

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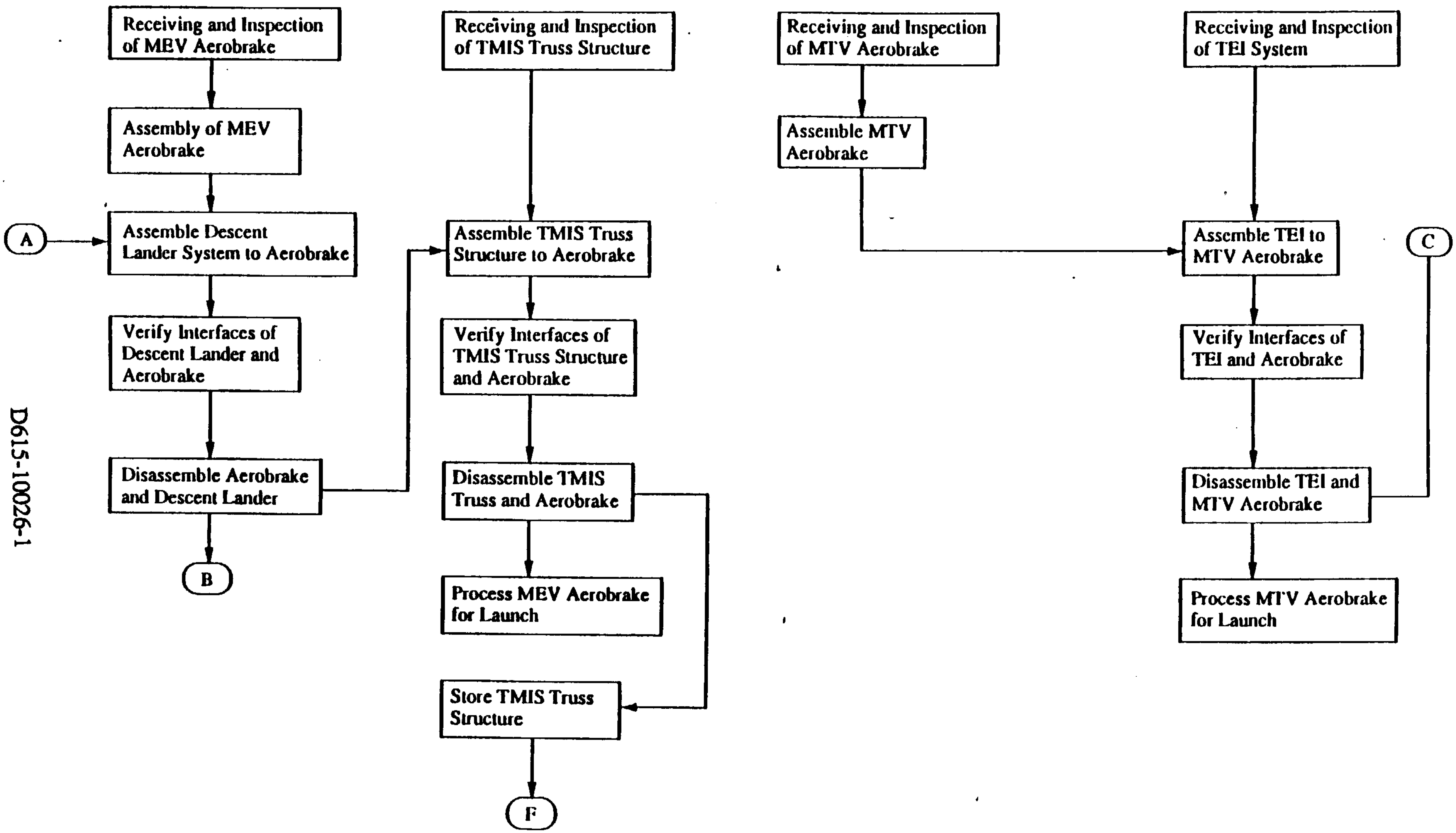
Ground Processing Functional Flow



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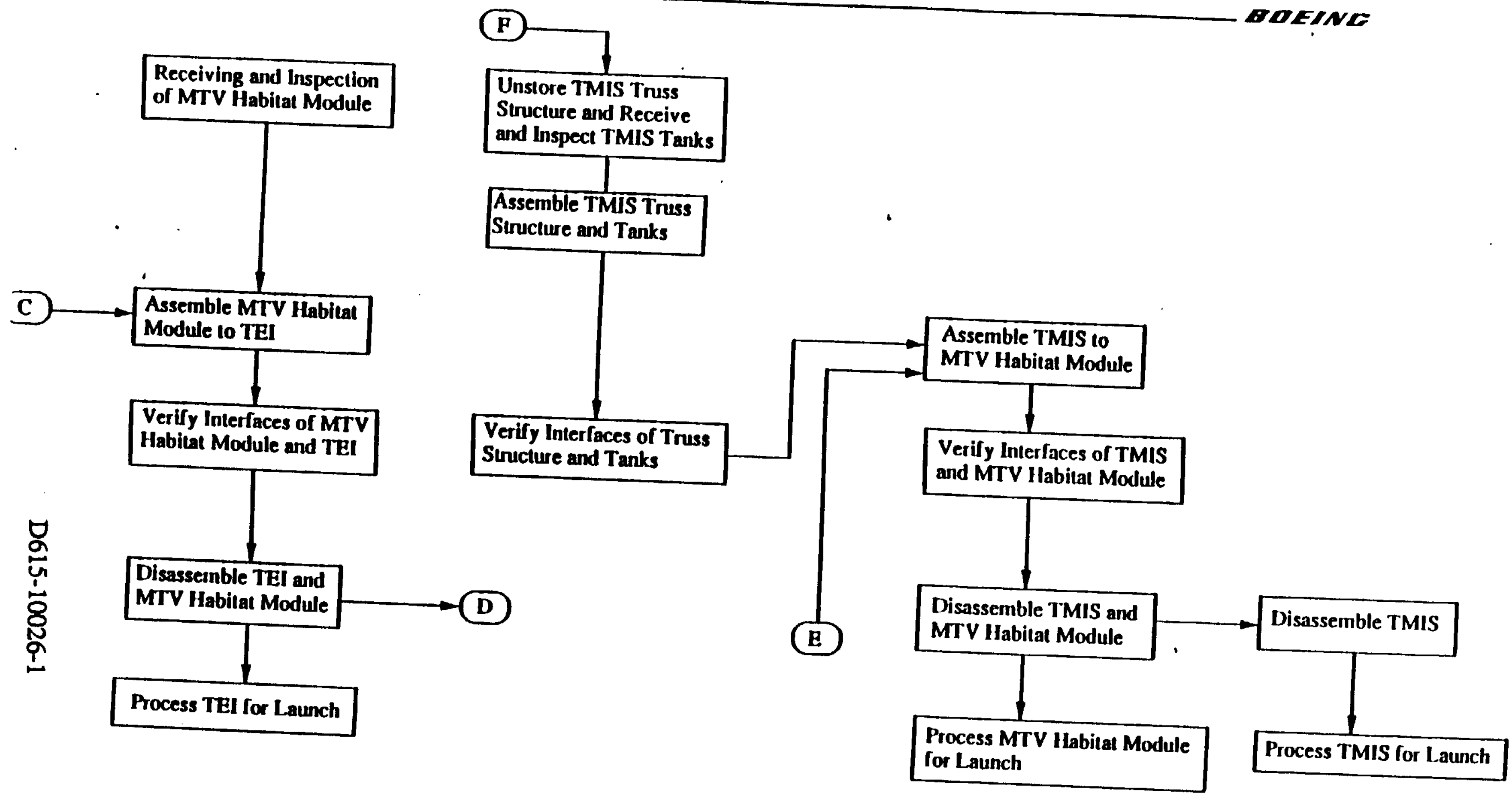
Ground Processing Functional Flow



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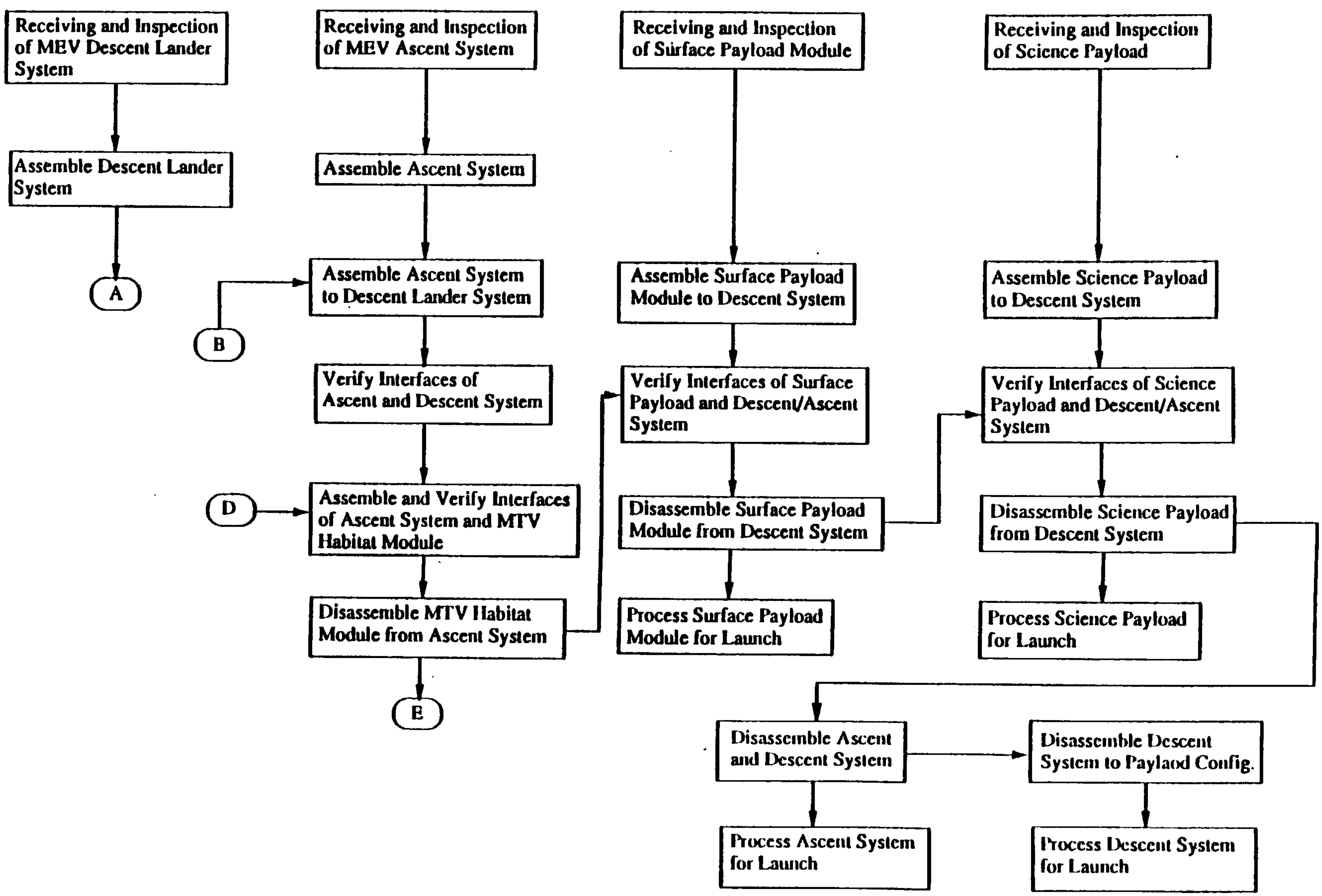

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Cryo/Aerobrake Test Philosophy

The following chart is the summary of the Test Philosophy developed for this analysis. The complete Philosophy and Approach was included in the May Progress Report.

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Test Philosophy

Purpose: Establish criteria and overall test approach that verifies a system is flight ready and will accomplish its mission successfully

Goals:

- Reduce redundant testing
- Reduce man power requirements for ground processing
- Reduce overall cost
- Provide system operational history

Criteria:

- Self test software and peripheral equipment will perform mechanical, electrical and electronic system tests and readiness analysis in an autonomous fashion
- Redundant flight hardware (on-board systems) will be under continuous self check
- Physical interfaces will be self latching connectors
- Prototype systems will be utilized when feasible for ground processing activities
- Commonality of systems will be stressed

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Ground Processing Facility Requirements

The following chart is a preliminary analysis of the Facility Requirements for Ground Processing of the Cryo/Aerobrake Vehicle.

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Facility Requirements

- **Processing Facility Ground Rules:**
  - Utilize Standard Services: Cranes, Power, Communications, Clean Rooms, etc.
  - Make Unique Hardware Portable: Special Test Equipment, Work Stations, Handling Fixtures, etc.
  - Provide Large Volume Workspace that can be readily adapted to System Block changes, Multiple Systems in Flow.
  - Provide for Hazardous System Processing

- **Equipment Requirements**
  - Overhead Crane
  - Flat Floor / Air Pallets
  - Standard Commerical Power
  - Uninterrupted Instrumentation Power
  - Environmental Control System: Humidity 50 +/- 5%   Temperature 75 +/- 5F
  - 100K Cleanliness Level
  - Closed Circuit Television
  - Facility GN2
  - Helium Supply
  - Shop Air
  - Fire Protection / Deluge
  - Shower / Eye Wash
  - Vacuum
  - Lightning Protection
  - Potable Water
  - Paging
  - Commerical Telephone
  - RF System
  - Operational Intercom System
  - Personnel Airlock
  - Grounding
  - Transportation/Ground Handling Fixtures

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Revised On-Orbit Analysis

The following two charts summarize the results of the Revised On-Orbit Assembly Analysis. Revisions were made because the analysis for HLLV processing was found to be in series to On-Orbit Assembly, where as the analysis should have been in parallel.

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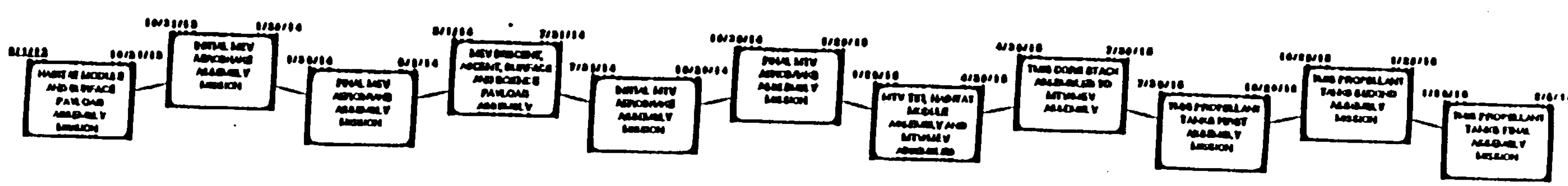
On-Orbit Assembly Analysis

- Assembly Analysis Presented at Second Quarter Review has Been Revised
- Revised Data shows 5 months less time required for On-Orbit Assembly
  - Original HLLV ground processing time was calculated in series with the On-Orbit Assembly time
  - Revised analysis calculates the HLLV ground processing time in parallel with the On-Orbit Assembly time
- Original On-Orbit Assembly Start date of January 2014
- Original On-Orbit Assembly Completion date of December 2016
- Revised On-Orbit Assembly Start date of August 2013
- Revised On-Orbit Assembly Completion date of February 2016

REVISED ASSEMBLY SCHEDULE



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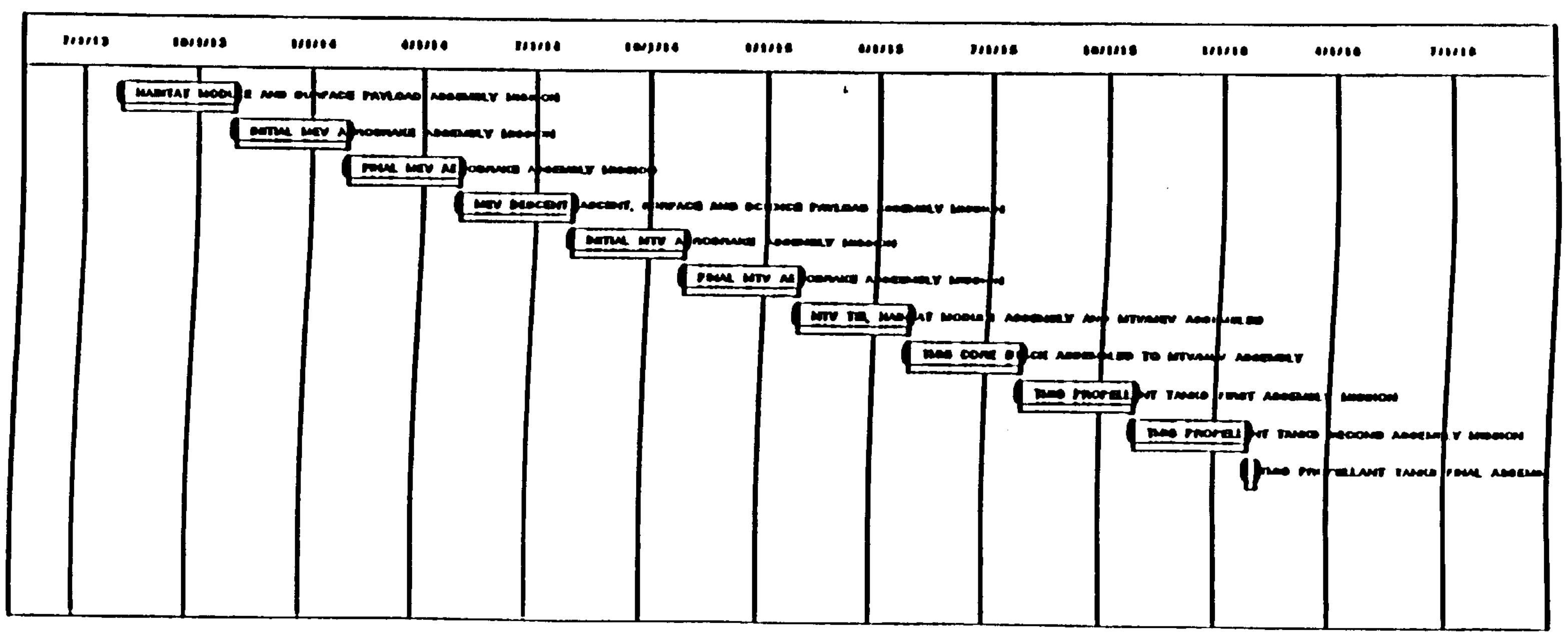
On-Orbit Assembly Analysis

Name	Earliest Start	Earliest Finish	Subproject	Days
HABITAT MODULE AND SURFACE PAYLOAD ASSEMBLY MISSION	8/1/13	10/31/13	HLLV MISSION ONE	91
INITIAL MEV AEROBRAKE ASSEMBLY MISSION	10/31/13	1/30/14	HLLV MISSION TWO	91
FINAL MEV AEROBRAKE ASSEMBLY MISSION	1/30/14	5/1/14	HLLV MISSION THREE	91
MEV DESCENT, ASCENT, SURFACE AND SCIENCE PAYLOAD ASSEMBLY	5/1/14	7/31/14	HLLV MISSION FOUR	91
INITIAL MTV AEROBRAKE ASSEMBLY MISSION	7/31/14	10/30/14	HLLV MISSION FIVE	91
FINAL MTV AEROBRAKE ASSEEMBLY MISSION	10/30/14	1/29/15	HLLV MISSION SIX	91
MTV TEI, HABITAT MODULE ASSEMBLY AND MTV/MEV ASSEMBLED	1/29/15	4/30/15	HLLV MISSION SEVEN	91
TMIS CORE STACK ASSEMBLED TO MTV/MEV ASSEMBLY	4/30/15	7/30/15	HLLV MISSION EIGHT	91
TMIS PROPELLANT TANKS FIRST ASSEMBLY MISSION	7/30/15	10/29/15	HLLV MISSION NINE	91
TMIS PROPELLANT TANKS SECOND ASSEMBLY MISSION	10/29/15	1/28/16	HLLV MISSION TEN	91
TMIS PROPELLANT TANKS FINAL ASSEMBLY MISSION	1/28/16	2/5/16	HLLV MISSION ELEVEN	8



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Impact on Diagnostics and Spares

An upgrade in the quality and speed of diagnostic capability will be needed to support the self-check , autonomous health monitoring with an assist database, and model based reference systems. All of these must be reliable, fault tolerant systems with high speed networks. They will be called upon to keep up with events that happen faster than human response time. They must be capable of knowing the difference between a sensor failure and a system failure.

The missions to Mars are measured in years, all supplies including spares must be determined prior to mission start. Systems must be as common as possible at the component level, in case an emergency dictates the cannibalization of auxiliary systems to allow life critical systems to operate. On-board repair capability will be required for the crew to both maintain the systems under normal conditions and have the ability to employ work arounds, functional repairs, and augment systems in the face of unforeseen or emergency conditions.

Impact on Diagnostics and Spares

Diagnostics:

- System ability to isolate a sensor failure from a system failure
- Systems must check pathways and operations
- Autonomous vehicle health monitoring
- Expert systems and assist database required
- On-board computer model -based reference system required
- High speed neural net programs and computers required
- High precision sensors with fault tolerant systems

Spares:

- Commonality of similar parts/systems is a high priority
- 15% of active component weight is allocated to spares
- Mean-time-between-failures (MTBF) will be measured in years
- Two complete and separate systems for life critical systems and spares are required
- Access to all major components for changeout is required
- Changeout, replacement and repair must be at the "tool kit" level

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Earth-to-Orbit (ETO) Heavy Lift Launch Vehicle (HLLV) Definition Trades.

An airplane must be designed considering the airports it flies to and from and the cargo/passengers it will have to carry. In a similar fashion a launch vehicle must be designed considering cargo manifest and transportation node requirements.

Externally mounted aerobrake. Payload fairings are used on launch vehicles when the payload cannot withstand the aerodynamic and heating environment of launch. The Space Shuttle Orbiter is an example of an item which is rugged enough to be launched without a protective fairing. Since an aerobrake of the type being considered for SEI missions is designed to operate under severe aerothermal conditions, consideration was given to launching it without a fairing.

Aerobrakes are large and low density. If the brakes are launched with a protective fairing, reasonable sizes of launch vehicles leads to volume-limited launches of brake segments. This adds launches and requires on-orbit assembly. The latter can add weight to the aerobrake in the form of field joints for the brake segments and also requires a 'final assembly building' in orbit. The potential penalties of launching an exposed aerobrake include (1) ascent performance penalties to the launch vehicle, and (2) structural and thermal protection additions if the launch environment is different than the brake mission environment.

The first two charts illustrate some configurations of an aerobrake on a Shuttle-derived launch vehicle. Launch performance is being investigated to determine feasibility of this concept. If it proves feasible, then the next step would to be to compare the brake design impact for external launch to the impact for launch in segments.

HLLV Sizing. The optimal size of a HLLV is a tradeoff between increased development cost for a larger vehicle versus fewer flights and lower on-orbit assembly costs. The next chart shows the variation in number of launches and achieved payload average mass for three sizes of HLLV and four transfer vehicle propulsion options under consideration.

Transport Node Location. The next chart gives comparative data on alternative locations for a transportation node. From a propulsive standpoint, a transport node should be located as close to the Earth as possible if a shift from low Isp (chemical rocket) to high Isp (electric) is occurring, so as to maximize the fraction of the mission ΔV performed by the higher Isp propulsion. Even if the Isp is not changing (chemical rocket both for ETO and transfer), the closer the node is to the Earth, the less propellant expended to get the node hardware on site.

Manifesting. The remaining charts in this section deal with manifesting of a ETO HLLV. Two large fairings are considered: a 7.6x30m fairing with a 120 ton lift capacity, and a 10x30m fairing with an 84 ton capacity. A nuclear electric propulsion (NEP) vehicle will require seven ETO launches. This includes two launches with the smaller fairing and five with the larger. A solar electric propulsion (SEP) vehicle requires only six ETO launches, with five of them using the larger fairing. A nuclear thermal rocket (NTR) transfer vehicle will require nine ETO launches, with six required to be the larger fairing size. The implied vehicle design density for payload envelopes is about 40 kg payload per cubic meter of volume. This means the vehicle should be optimized for this payload density, with allowance for putting larger or smaller fairings on for particularly dense or bulky payloads.

Using a larger ETO Heavy Lift Launch Vehicle, with a 10x30m fairing and a 140 t capacity reduces the number of launches required. In the NEP case the launches are reduced from

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seven to five. In the SEP case the flights are reduced from six to five. In the NTR case the launches are reduced from nine to seven, although one flight is very lightly loaded.

The last several charts compare a cryogenic propulsion transfer vehicle with aerobrakes for arrival at planets to other transfer vehicles as far as ETO flights required. In the mixed 7.6x30/10x30m HLLV case, 11 flights are required for the Cryogenic transfer vehicle, vs 9,6, and 7 for NTR, SEP, and NEP respectively. For the larger, 140 metric ton capacity HLLV, 8 launches are required for the Cryogenic transfer vehicle, versus 7, 5, and 5 for the NTR, SEP and NEP respectively.

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Integrated Aerobrake Launch Option

The following chart shows a Shuttle-C / Aerobrake Integrated Launch Option. This option would launch a single Aerobrake to LEO along with other payload stored in the Shuttle-C Payload Shroud.

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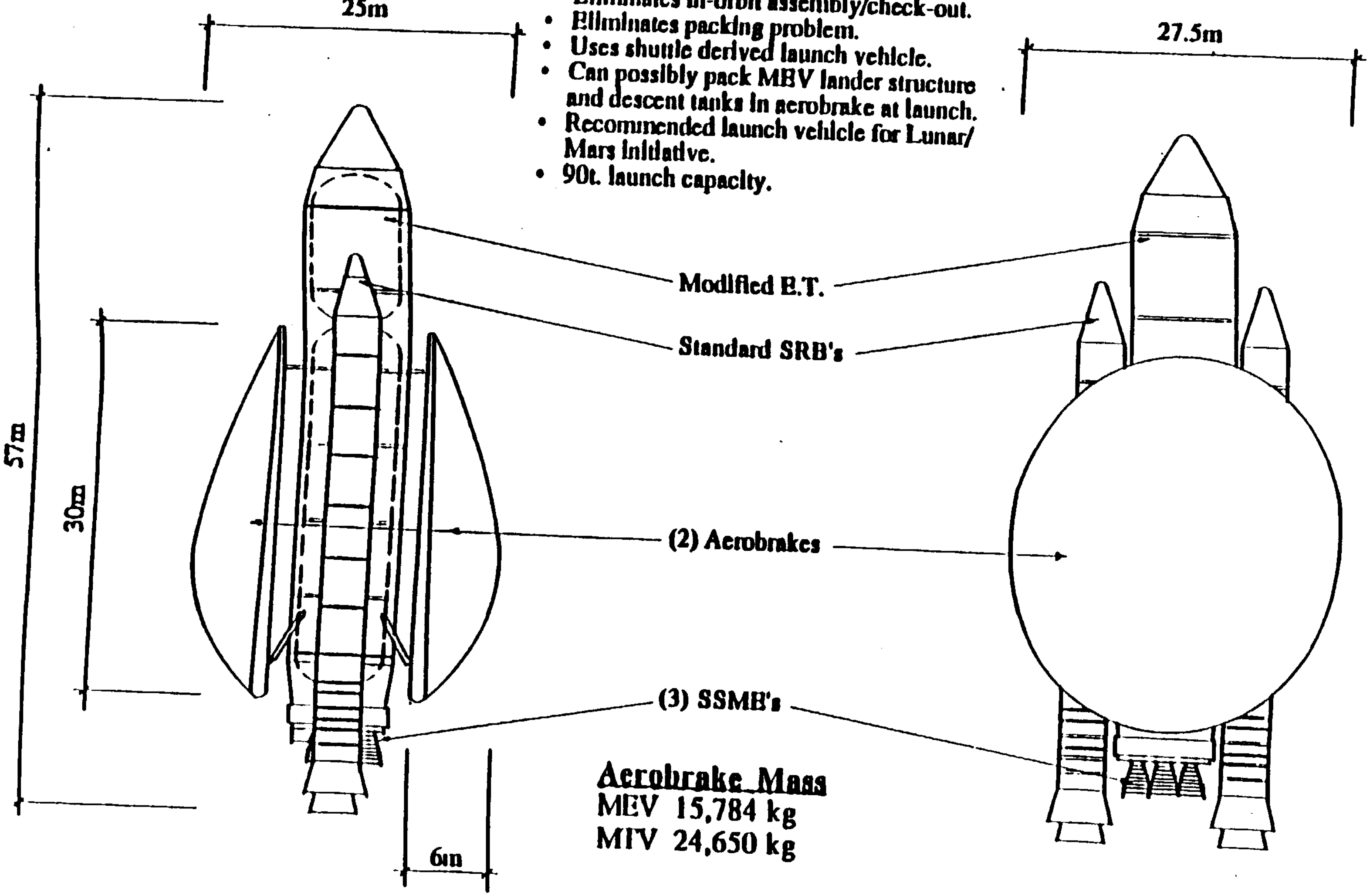
Shuttle Derived Aerobrake Launch Option

- Aerobrake completely assembled on Earth.
- Eliminates in-orbit assembly/check-out.
- Eliminates packing problem.
- Uses shuttle derived launch vehicle.
- Can possibly pack MEV lander structure and descent tanks in aerobrake at launch.
- Recommended launch vehicle for Lunar/Mars Initiative.
- 90t. launch capacity.



Aerobrake Mass
MEV 15,784 kg
MTV 24,650 kg

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Integrated Aerobrake Launch Option

On-Orbit Assembly of the Mars Aerobrake(s) require two 10.5 meter dia HLLV launches each and 180 days (due to 90 day ground processing time required for the HLLV). A concept which would deliver an assembled aerobrake or aerobrakes to LEO is shown in the following chart. This concept utilizes a Shuttle derived In-Line vehicle to launch two aerobrakes to LEO.

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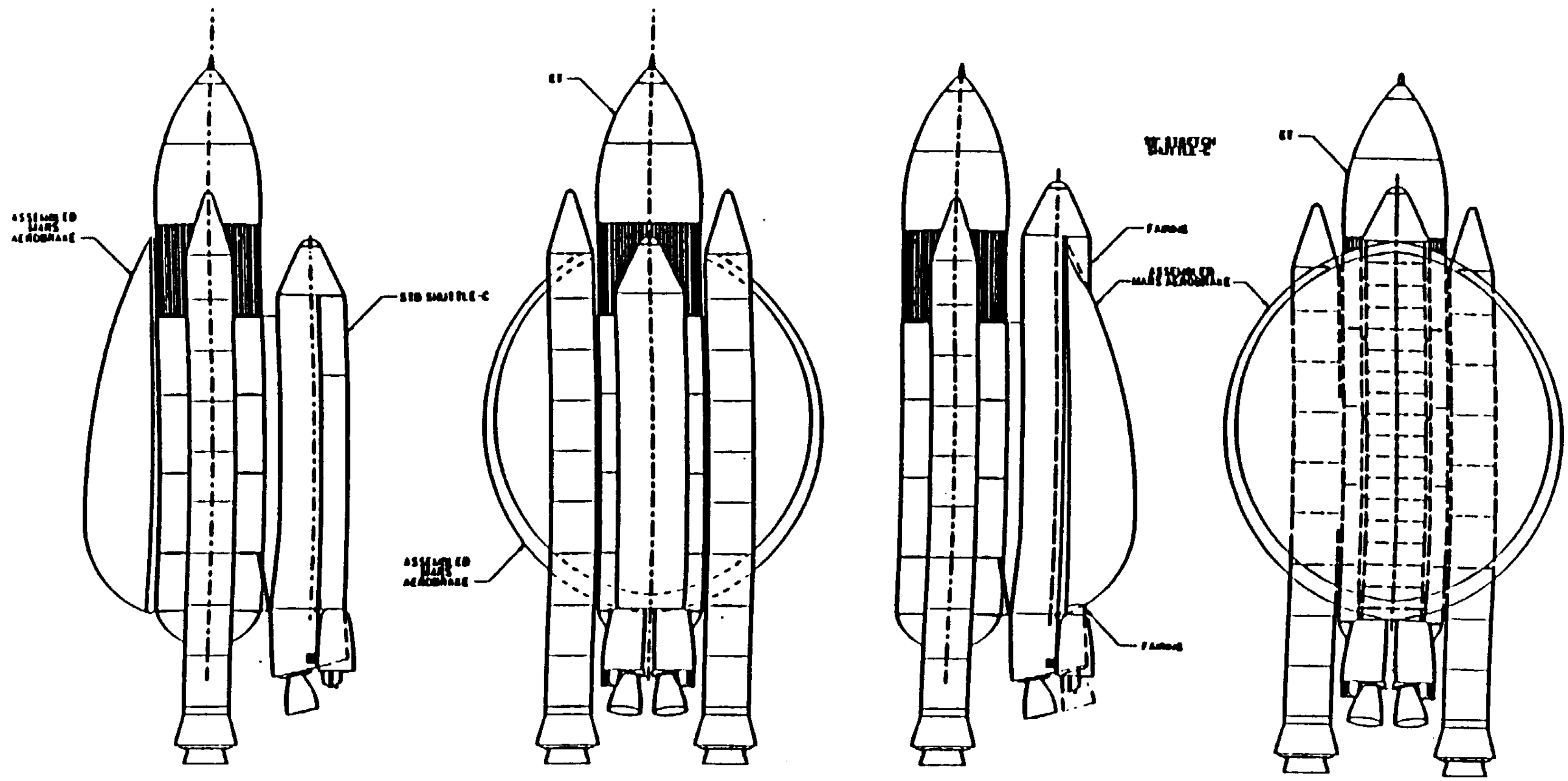
Shuttle-C Aerobrake Launch Options



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HLLV Optional Manifesting

Optional Manifesting of the four vehicle options was developed for a medium and large class HLLV. The analysis was completed by using theoretical volumetric and total mass calculations.

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HLLV Optional Manifesting

- Manifesting data will be generated by volumetric and mass total calculations
- Aerobrake(s) will be assembled on-orbit
- Deployable truss type mechanisms are feasible
- Manifesting assumed on-orbit assembly at LEO

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On-Orbit Assembly Analysis (HLLV Missions to LEO)

Proplusion Option / ETO Size	10 meter Dia. 30 meter long 84-120 mT class	12.5 meter Dia. 30 meter long 140-160 mT class	13.75 meter Dia. 38 meter long/22 meter nose cone 200-250 mT class
Cryo/Aerobrake (opposition class)	11 missions 73 mT average	8 missions 101 mT average	5 missions 162 mT average
Solar Electric Power (opposition class)	TBD	5 missions 87.2 mT average	2 missions 218 mT average
Nuclear Electric Power (opposition class)	TBD	5 missions 108.9 mT average	3 missions 181.5 mT average
Nuclear Thermal Rocket (opposition class)	TBD	6 missions 122 mT average	4 missions 183 mT average

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Assembly Node Purpose and Requirements

The Assembly Node Purpose and Node Requirements were developed to support a study that MSFC was performing to determine the advantages and disadvantages for different orbits and assembly node. Each vehicle option has advantages for being assembled and launched from different orbits. The data shown in the following charts are for the Cryo/Aerobrake Vehicle.

Node Purpose & Minimal Requirements

Purpose	• **Integration** • **De-integration** • **Re-integration** of mission vehicles, including assembly, processing, resupply and refurbishment
Top-level Requirements	• **Accessibility** • **Provision of support services to mission vehicle**
Functional Approach	• **Analyze specific functions necessary to provide required services to the mission vehicles** • **Identify synergistic ways of providing those functions, emphasizing "operational" solutions (e.g. using/proving onboard vehicle systems, resupplying before mission departure)** • **Defer device-driven solutions until minimum common requirements are distilled, which cannot be satisfied by hardware already "procured" for the vehicle itself**

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Node Comparisons

Node / Data to Mars	LEO MASE REF.	GEO Case 1	GEO* Case 3	LLO*	L2*
Δ V to Mars (m/sec)	4281	5402	2423	1995	1315
TMI S mass	470.3 t	751.5 t	191.8 t	147.7 t	87.9 t
Debris Env.(total time)	Continuous	0	< 1 hr	< 1 hr	< 1 hr
Radiation Env. total time in Van Allen belts	~2hrs	0	< 2 hrs	< 2 hrs	< 2 hrs
Radiation Environment	Trapped, SAA	Trapped, GCR, SPE	Trapped, GCR, SPE	GCR, SPE	GCR, SPE
Crew to Node ΔV /t	SSF*** ΔV=10-100m/sec hrs-days	SSF - 6 hrs ΔV= 4200m/sec	SSF - 6 hrs ΔV=4200m/sec	SSF- 3-13days ΔV= 4000 m/s Moon- 2hr ΔV= 2100m/s	SSF -8-18 days ΔV= 3374m/s Moon -3 days ΔV= 2900m/s
Launch window timing	e.s o.** ~ 10 min orbit align ~ 5d opps/day ~15-16 rec = 30-60 days	e.s.o.~ 1/day orbit align ~ planetary position opp ops/day = continous		e.s.o. @ 10 min alignment =12hr rec.= 27 days retrograde opp.	alignment =12hr rec.= 27 days retrograde opp.
Logistics wndow timing	anytime	5 hr transit, 2 opportunities/day		every 10 days	transit from Moon = 3 day

* denotes PEGA (powered Earth gravity assist)
** engine start opportunity
*** co- orbiting with SSF costs ΔV to maintain poisition relative to SSF(not continous thrust)

GCR = Galatic Cosmic Radiation
SPE = Solar Proton Events

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ADVANCED CIVIL SPACE SYSTEMS

MIXED LAUNCH FLEET MANIFESTING ASSUMPTIONS

BOEING

- Available Heavy Lift Launch Vehicles (HLLVs) include 7.6m x 30m shroud (120mt) and 10m x 30m shroud (84mt) with some TBD volume available in nosecone
- Nominal 0.5 meter clearance required between payloads and between payload and shroud to account for undefined OSE/FSE (violations do occur and are assumed workable)
- No CG constraints (payload or HLLV) accounted for other than to locate launch CG aft (by placing heavier payloads at the bottom of stack)
- NEP/SEP/NTR configuration (volume and mass) current as of 8/15/90 (with 8/30/90 change to SEP arrays)
- Off-SSF Assembly Node available and functional for assembly ops (manned and robotic), maintenance, and launch (4 assembly crew are assumed to be SSF-based; a POTV is available and capable of crew transfer (both pressurized and EVA) as well as equipment transfer/manipulation)
- No artificial gravity capability
- MEV propellant/engines, structure, airlock, plumbing/cabling (within aerobrakes, etc.), tunnels, rovers, power arrays and radiators (for non-engine systems during transfer and excursion), etc. assumed to be included (or TBD)
- MEV ascent/descent stage propellant/tanks, engines, structure, etc. assumed to be included with module (or TBD)
- Transfer vehicle robotics and some Assembly Node systems part of "Assembly Equipment"
- MTV to be brought up as late in sequence as possible to support ground integration/verification
- Boiloff is assumed controllable; on-orbit propellant transfer assumed workable
- "Ninja Turtle" aerobrake launch concept assumed baseline (no impact to HLLV payload capacity assumed)

ADVANCED CIVIL SPACE SYSTEMS

NTR MANIFESTING USING MIXED LAUNCH FLEET
(Sheet 1 of 2)

BOEING

	Mass (metric tons)
Mission One:*	
Descent Module	32.83
Ascent Module	24.83
Surface Payload	25.0
Aerobrake (Externally Mounted)	9.51
Assembly Equipment	20.0
	112.17
Mission Two:**	
Mars Departure/Earth Arrival Structure	1.0
Main Truss	5.3
MTV Hab	40.3
	46.6
Mission Three:**	
Engine/Shield Structure	2.4
Mars Departure/Earth Arrival Tank (13.5 mt offloaded)	81.6
	84.0
Mission Four:**	
Mars Arrival/Departure Tank (1 of 2)	82.9

* 7.6m x 30m Shroud (120 mt)
** 10m x 30m Shroud (84 mt)

17sep90/ter/manifesting/ninja turtle

NTR MANIFESTING USING MIXED LAUNCH FLEET
(Sheet 2 of 2)

	Mass (metric tons)
Mission Five:**	
Mars Arrival/Departure Tank (Remaining 1 of 2)	82.9
Mission Six:**	
Earth Departure Tank (1 of 2, 69.3 mt offloaded)	84.0
Mission Seven:**	
Earth Departure Tank (Remaining 1 of 2, 69.3 mt offloaded)	84.0
Mission Eight:***	
Offloaded Propellant from Missions 3 and 6	79.5
Mission Nine:***	
Reactor Engine and Shield	11.4
Offloaded Propellant from Missions 6 and 7	72.6
	84.0

* 7.6m x 30m Shroud (120 mt)
** 10m x 30m Shroud (84 mt)
*** May be either HLLV

NEP MANIFESTING USING MIXED LAUNCH FLEET
(Sheet 1 of 2)

	Mass (metric tons)
Mission One:*	
Descent Module	32.83
Ascent Module	24.83
Surface Payload	25.0
Aerobrake (Externally Mounted)	9.51
Assembly Equipment	20.0
	112.17
Mission Two:**	
Power Distribution and Control	32.8
Structure (2 deployables 7x7x7m)	4.6
Attitude Control	5.7
Power Conditioning	1.8
Communications	0.6
Avionics	2.5
	48.0
Mission Three:**	
Propellant and Tanks (1 of 5)	37.0
Thruster Pods (3 of 4)	41.1
	78.1

* 7.6m x 30m Shroud (120 mt)
** 10m x 30m Shroud (84 mt)

NEP MANIFESTING USING MIXED LAUNCH FLEET
(Sheet 2 of 2)

	Mass (metric tons)
Mission Four:**	
Main Cycle Radiators (2 of 2)	10.7
Thruster Pod (4th of 4)	13.7
Propellant and Tanks (2nd of 5)	37.0
	61.4
Mission Five:**	
Auxiliary Radiators (2 of 2)	5.1
Turbopumps/Heat Transport Loops/etc.	71.5
	76.6
Mission Six:**	
Propellant and Tanks (3rd of 5)	37.0
MTV Hab	40.3
	77.3
Mission Seven:*	
Reactors and Shields	27.4
Propellant and Tanks (Remaining 2 of 5)	74.0
	101.4

* 7.6m x 30m Shroud (120 mt)
** 10m x 30m Shroud (84 mt)

SEP MANIFESTING USING MIXED LAUNCH FLEET
(Sheet 1 of 2)

	Mass (metric tons)
Mission One:*	
Descent Module	32.83
Ascent Module	24.83
Surface Payload	25.0
Aerobrake (Externally Mounted)	9.51
Assembly Equipment	20.0
	112.17
Mission Two:**	
Array Deployment Mechanism	2.6
Communications	0.6
Attitude Control	5.7
Avionics	1.7
Power Processing Units	4.7
Power Distribution & Control	8.2
Main Truss (Deployable 7x7x7m, 2 of 3)	6.13
Array Structure (Deployable 7x7x7m, 2 of 2)	3.6
Experimental Platforms	1.7
	34.93
Mission Three:**	
Propellant and Tanks (1 of 4)	38.5
Radiators (2 of 2)	6.9
Main Truss (Deployable 7x7x7m, remaining 1 of 3)	3.07
Transfer Arrays (10 of 18)	14.44
	62.91

*** 7.6m x 30m Shroud (120 mt)**
**** 10m x 30m Shroud (84 mt)**

17sep90/tcr/manifesting/mmp.tmh

SEP MANIFESTING USING MIXED LAUNCH FLEET
(Sheet 2 of 2)

	Mass (metric tons)
Mission Four:**	
Propellant and Tanks (2nd of 4)	38.5
Transfer Arrays (Remaining 8 of 18)	11.56
Thruster Pods (2 of 2)	13.7
	63.76
Mission Five:**	
Propellant and Tanks (3rd of 4)	38.5
Array Blankets (15 of 18)	21.83
	60.33
Mission Six:**	
Propellant and Tanks (Remaining 1 of 4)	38.5
Array Blankets (Remaining 3 of 18)	4.37
MTV Hab	40.3
	83.17

* 7.6m x 30m Shroud (120 mt)
** 10m x 30m Shroud (84 mt)

17sep90/ter/manifesting/ninja turtle

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10m x 30m SHROUD, 140 mt HLLV MANIFESTING ASSUMPTIONS

- Heavy Lift Launch Vehicle (HLLV) with 140 metric ton payload capability and 10m x 30m shroud
- Volume is available in nosecone of HLLV
- Nominal 0.5 meter clearance required between payloads and between payload and shroud (violations do occur and are assumed workable)
- No CG constraints (payload or HLLV) accounted for other than to locate launch CG aft (by placing heavier payloads at the bottom of stack)
- FSE/OSE not defined
- NEP/SEP/NTR configuration (volume and mass) current as of 8/15/90 (with 8/30/90 change to SEP arrays)
- Off-SSF Assembly Node available and functional for assembly ops (manned and robotic), maintenance, and launch
- No artificial gravity capability
- MEV propellant/engines, structure, airlock, plumbing/cabling (within aerobrakes, etc.), tunnels, rovers, power arrays and radiators (for non-engine systems during transfer and excursion), etc. assumed to be included (or TBD)
- MEV ascent/descent stage propellant/tanks, engines, structure, etc. assumed to be included with module (or TBD)
- Transfer vehicle robotics and Assembly Node systems part of "Assembly Equipment"
- Manifested volumes are assumed to account for protuberances, etc.
- Boiloff is assumed controllable; on-orbit propellant transfer assumed workable
- "Ninja Turtle" aerobrake launch concept assumed baseline

NTR MANIFESTING USING 10m x 30m SHROUD, 140 mt HLLV
(Sheet 1 of 2)

	Mass (metric tons)
Mission One:	
MTV Hab (Module and systems)	40.3
Ascent Module	24.83
Descent Module	32.83
Main Truss (Deployable 7x7x7m)	5.3
Aerobrake (Externally mounted)	9.51
Assembly Equipment	20.0
	132.77
Mission Two:	
Mars Dept./EA Structure	1.0
Surface Payload	25.0
Mars Dept./EA Tank	95.1
	121.1
Mission Three:	
Earth Departure Tank (1 of 2, 13.3 mt offloaded)	140.0
Mission Four:	
Earth Departure Tank (2nd of 2, 13.3 mt offloaded)	140.0

NTR MANIFESTING USING 10m x 30m SHROUD, 140 mt HLLV
(Sheet 2 of 2)

	Mass (metric tons)
Mission Five:	
Engine/Shield Structure	2.4
Mars Arrival/Dept. Tank (1 of 2)	82.9
	85.3
Mission Six:	
Reactor Engine and Shield	11.4
Mars Arrival/Dept. Tank (2nd of 2)	82.9
	94.3
Mission Seven:*	
Offloaded Propellant from Missions 3 and 4	26.6

* May be STS-derived launch vehicle

30aug91/vlci/manifesting/ompa turtle

NEP MANIFESTING USING 10m x 30m SHROUD, 140 mt HLLV
(Sheet 1 of 2)

ADVANCED CIVIL SPACE SYSTEMS — BOEING

	Mass (metric tons)
Mission One:	
MTV Hab (Modules and systems)	40.3
Structure (2 deployables 7x7x7m)	4.6
Attitude Control	5.7
Power Conditioning	1.8
Communications	0.6
Avionics	2.5
Aerobrake (Externally mounted)	9.51
Assembly Equipment	20.0
	85.01
Mission Two:	
Descent Module	32.83
Ascent Module	24.83
Surface Payload	25.0
Auxiliary Radiators (1 of 2)	2.55
Power Distribution & Control	32.8
	118.01
Mission Three:	
Propellant and Tanks (2 of 5)	74.0
Thruster Pods (3 of 4)	41.1
	115.1

NEP MANIFESTING USING 10m x 30m SHROUD; 140 mt HLLV
(Sheet 2 of 2)

	Mass (metric tons)
Mission Four:	
Main Cycle Radiators (1 of 2)	5.35
Thruster Pod (4th of 4)	13.7
Propellant and Tanks (Remaining 3 of 5)	111.0
Auxiliary Radiators (2nd of 2)	2.55
	132.6
Mission Five:	
Main Cycle Radiators (2nd of 2)	5.35
Reactors/Shield/Turbopumps/etc.	98.9
	104.25

30aug90/ter/manifesting/nmja turtle

ADVANCED CIVIL SPACE SYSTEMS

SEP MANIFESTING USING 10m x 30m SHROUD, 140 mt HLLV
(Sheet 1 of 2)

BOEING

	Mass (metric tons)
Mission One:	
MTV Hab (Module and systems)	
Array Deployment Mechanism	40.3
Communications	2.6
Attitude Control	0.6
Avionics	5.7
Power Processing Units	1.7
Power Distribution & Control	4.7
Main Truss (Deployable 7x7x7m, 1 of 3)	8.2
Array Structure (Deployable 7x7x7m, 2 of 2)	3.1
Experimental Platforms	3.6
Assembly Equipment (1 of 2)	1.7
	10.0
	82.2
Mission Two:	
Transfer Arrays (18 of 18)	
Aerobrake (Externally Mounted)	26.0
Assembly Equipment (2nd of 2)	9.51
	10.0
	45.51
Mission Three:	
Descent Module	
Ascent Module	32.83
Surface Payload	24.83
Main Truss (Deployable 7x7x7m, remaining 2 of 3)	25.0
	6.13
	88.8

30aug90/tcr/manifesting/ninja turtle

SEP MANIFESTING USING 10m x 30m SHROUD, 140 mt HLLV
(Sheet 2 of 2)

	Mass (metric tons)
Mission Four:	
Propellant and Tanks (2 of 4)	77.0
Radiators	6.9
Array Blankets (14 of 18)	20.38
	104.28
Mission Five:	
Propellant and Tanks (Remaining 2 of 4)	77.0
Thruster Pods	13.7
Array Blankets (Remaining 4 of 18)	5.82
	96.52

Mars Mission Manifests-
Cryo/aerobrake and NTR
84 -120 t HLLV

HLLV 1 : Shroud Size- 30 meters x 10 m dia., 84 t throw weight
HLLV 2 : Shroud Size - 30 meters x 7.6 m dia., 120 t throw weight

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Cryo/Aerobrake

Flight #	1	2	3	4	5	6	7-10	11
Launch Vehicle	HLLV 1	HLLV 1	HLLV 1	HLLV 1	HLLV1	HLLV 2	HLLV 2	HLLV 1
Manifest	MTV Hab Module, Surface Payload, Assembly Equipment	MEV Aerobrake (Ninja Turtle) Habitat refurbish/ consumables, Assembly Equipment	Descent Module, Ascent Module, structure	MTV Aerobrake (Ninja Turtle), Habitat refurbish/ consumables, Assembly Equipment, ECCV, TEI Tanks & Engines	TEIS Propellant, consumables	TMI Propellant & Engines	TMI Propellant and tanks	Top-off Propellant and equipment, Habitat refurbish/ consumables
Mass	78.8 t	40.1 t	49.1 t	76.4 t	84 t	117.7 t	114.5 t	84 t

NTR

Flight #	1	2	3	4	5	6	7	8	9
Launch Vehicle	HLLV 2	HLLV 1	HLLV 1	HLLV 1	HLLV1	HLLV 1	HLLV 1	HLLV 1 or 2	HLLV 1 or 2
Manifest	Descent Module, Ascent Module Surface Payload, MEV aerobrake (Ninja Turtle), Assembly Equipment	Mars Departure/ Earth Arrival Structure, Main Truss, MTV Hab	Engine/ Shield Structure, Mars Departure/ Earth Arrival tank (13.5 t off-loaded),	Mars Arrival/ Departure Tank (1 of 2)	Mars Arrival/ Departure Tank (2nd of 2)	Earth Departure Tank (1 of 2, 69.3 t off-loaded)	Earth Departure Tank (1 of 2, 69.3 t off-loaded)	Off-loaded Propellant from Flight 3 &6	Off-loaded Propellant from Flight 6 & 7, Reactor Engine and Shield
Mass	112.2 t	46.6 t	84.0 t	82.9 t	82.9 t	84 t	84 t	79.5 t	84 t

Mars Mission Manifests- SEP and NEP 84 - 120 t HLLV

ADVANCED CIVIL SPACE SYSTEMS

BOEING

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HLLV 1 : Shroud Size- 30 meters x 10 m dia., 84 t throw weight
HLLV 2 : Shroud Size - 30 meters x 7.6 m dia., 120 t throw weight

SEP

Flight #	1	2	3	4	5	6
Launch Vehicle	HLLV 2	HLLV 1	HLLV1	HLLV1	HLLV 1	HLLV1
Manifests	Descent Module, Ascent Module, Surface Payload, MEV Aerobrake (Ninja Turtle), Assembly Equipment	Array Deployment Mech., Communications, ACS PPUs, Avionics, Main Truss (3 of 4), Power Distribution & Control, Array Structure, Experiment Platforms	Propellant & Tanks (1 of 4), Radiators (2 of 2), Main Truss (4th of 4), Transfer Arrays (10 of 18)	Propellant & Tanks (2nd of 4), Transfer Arrays (8 of 18), Thruster Pods (2 of 2)	Propellant & Tanks (3rd of4), Array Blankets (15 of 18)	Propellant & Tanks (4th of 4), Array Blankets (last 3) MTV Hab.
Mass	112.2 t	34.9 t	62.9 t	63.8 t	60.3 t	83.2 t

NEP

Flight #	1	2	3	4	5	6	7
Launch Vehicle	HLLV 2	HLLV 1	HLLV 1	HLLV 1	HLLV 1	HLLV 1	HLLV 2
Manifests	Descent Module, Ascent Module, Surface Payload, MEV Aerobrake (Ninja Turtle), Assembly Equipment	Power Distribution and Control, Structure, ACS, Power Conditioning, Communications, Avionics	Propellant and Tanks (1 of 5), Thruster pods (3 0f 4)	Main Cycle Radiators (2 of 2), Thruster pod, Propellant and Tanks (2nd of 5)	Auxiliary Radiators (2 of 2), Turbopumps/ Thermal loops,	Propellant and Tanks (3rd of 5), MTV Hab,	Reactors & Shields, Propellant and Tanks (last 2)
Mass	112.2 t	48 t	78.1 t	61.4 t	76.6 t	77.3 t	101.4 t

ADVANCED CIVIL SPACE SYSTEMS

Mars Mission Manifests-
Cryo/aerobrake and NTR
140 t HLLV

BOEING

HLLV : Shroud Size- 30 meters x 10 m dia., 140 t throw weight

STCAEM/PB/9,15,90

Cryo/Aerobrake

Flight #	1	2	3	4	5	6	7	8
Launch Vehicle	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV
Manifest	MEV Aerobrake (Ninja Turtle), Descent stage, surface cargo, assembly equipment, consumables	Ascent stage, interconnect structure (MEV-MTV), TEI departure stage	MTV Hab Module, ECCV, MTV Aerobrake (Ninja Turtle), interconnect structure (MTV-TMIS), assembly equipment, consumables	TMIS tankage, Engine set, Structure, assembly equipment, Top-off equipment	TMIS tankage and propellant	TMIS tankage and propellant	TMIS tankage and propellant	TMIS tankage and propellant (wet tanks and top-off)
Mass	67.8 t	138.7 t	100 t	140 t	119.7 t	119.7 t	119.7 t	140 t

NTR

Flight #	1	2	3	4	5	6	7
Launch Vehicle	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV
Manifest	MEV Aerobrake (Ninja Turtle), MTV Hab, Ascent module, Descent module, Main truss, assembly equipment	Mars Departure Structure, Surface payload, Mars Departure Tankage	TMI Tank	TMI Tank	Engine, Shield structure, Mars Arrival/ Departure Tank	Reactor Engine and Shield, Mars Arrival/ Departure Tank	Propellant for top-off
Mass	132.8 t	121.1	140 t	140 t	85.3 t	94.3 t	26.6 t

ADVANCED CIVIL SPACE SYSTEMS

Mars Mission Manifests- SEP and NEP 140 t HLLV

BOEING

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HLLV : Shroud Size- 30 meters x 10 m dia., 140 t throw weight

SEP

Flight #	1	2	3	4	5
Launch Vehicle	HLLV	HLLV	HLLV	HLLV	HLLV
Manifest	MTV Hab, Array Deployment Mech., Communications, ACS, Avionics, PPUs, Power Distribution & Control, Main Truss, Array Structure, Experiment Platforms, Assembly Equipment	Transfer Arrays (18 of 18), MEV Aerobrake (Ninja Turtle), Assembly Equipment	Descent Module, Ascent Module, Surface Payload, Main Truss	Propellant Tanks, Propellant, Radiators, Array Blankets (14 of 18)	Propellant tanks, Propellant, Thruster pods, Array Blankets (4 of 18)
Mass	82.2 t	45.5 t	88.8 t	104.3 t	96.5 t

NEP

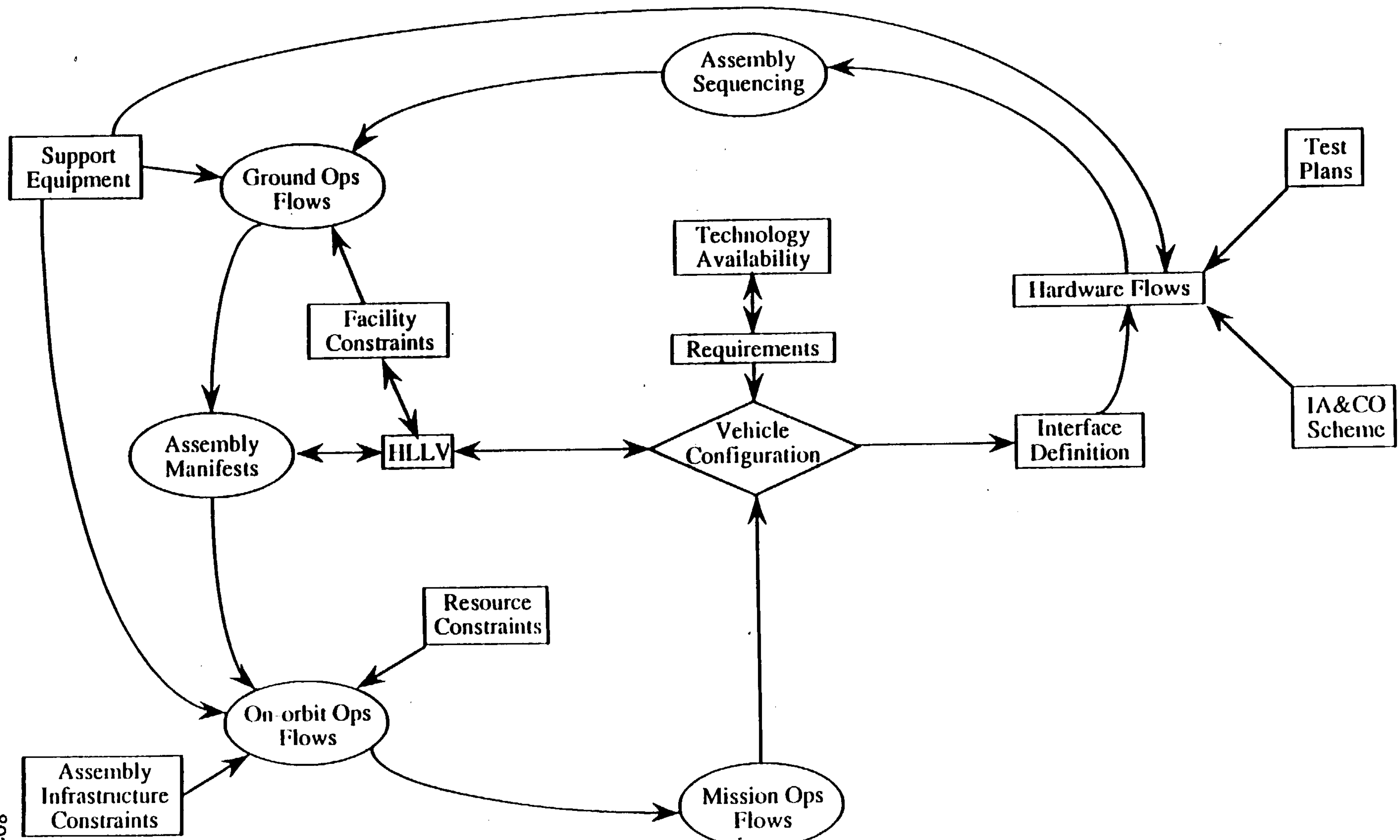
Flight #	1	2	3	4	5
Launch Vehicle	HLLV	HLLV	HLLV	HLLV	HLLV
Manifest	MTV Hab, MEV aerobrake (Ninja Turtle), ACS, Power conditioners, Communications, Assembly Equipment	Descent Module, Ascent Module, Surface Payload, Auxiliary Radiator, Power distribution and control	2 Propellant tanks, 3 thruster pods	1 Main Radiator, 1 thruster pod, 3 propellant tanks, 1 Auxiliary Radiator	2 nd Main Radiator, Reactors, Shields, Turbo pumps, Misc.
Mass	85.1 t	118.1 t	115.1 t	132.6 t	104.3 t

Manifesting and Assembly Operations

Vehicle configurations and each and every phase of assembly operations, is an iterative and interdependent process. As the vehicle evolves, in accordance with specific designed mission requirements, it must also conform to the existing and/or planned infrastructure. The vehicle configuration must reflect optimal ground processing and on-orbit assembly.

Manifesting and Assembly Operations

This is an iterative, interdependent process



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Manifesting and Assembly Operations- continued

Generic Assumptions and Ground Rules

- Based on Mars Vehicle (NEP, SEP, and NTR) configurations as of 3rd Quarterly Briefing with updates through 8/15/90
- Baseline Earth-to-Orbit (ETO) Vehicle (HLLV) has 10m x 30m shroud with 140 mt payload capability
- HLLV nosecone has some additional TBD volume for launch element packaging
- Nominal 85% payload packaging and mass factors used for HLLV manifesting (propellant tanks may be excepted)
- HLLV has a nominal 3 to 7 day station-keeping ability
- HLLV unloaded piece by piece by Cargo Transfer Vehicle (CTV)
- Crew transported to Assembly Location from SSF via ACRV
- CTV will be designed to support all identified manned/unmanned operations (on-orbit refueling may be available via on-orbit depot, HLLV provisioning, the Mars Vehicle itself, or SSF)
- HLLV launched on 90 day centers = time constraint for on-orbit assembly operations
- All Mars Vehicles are assumed to be launched February 2016
- Any localized debris shielding is removed from Mars Vehicle prior to departure from Earth (micrometeoroid shielding is assumed to be needed for the mission duration

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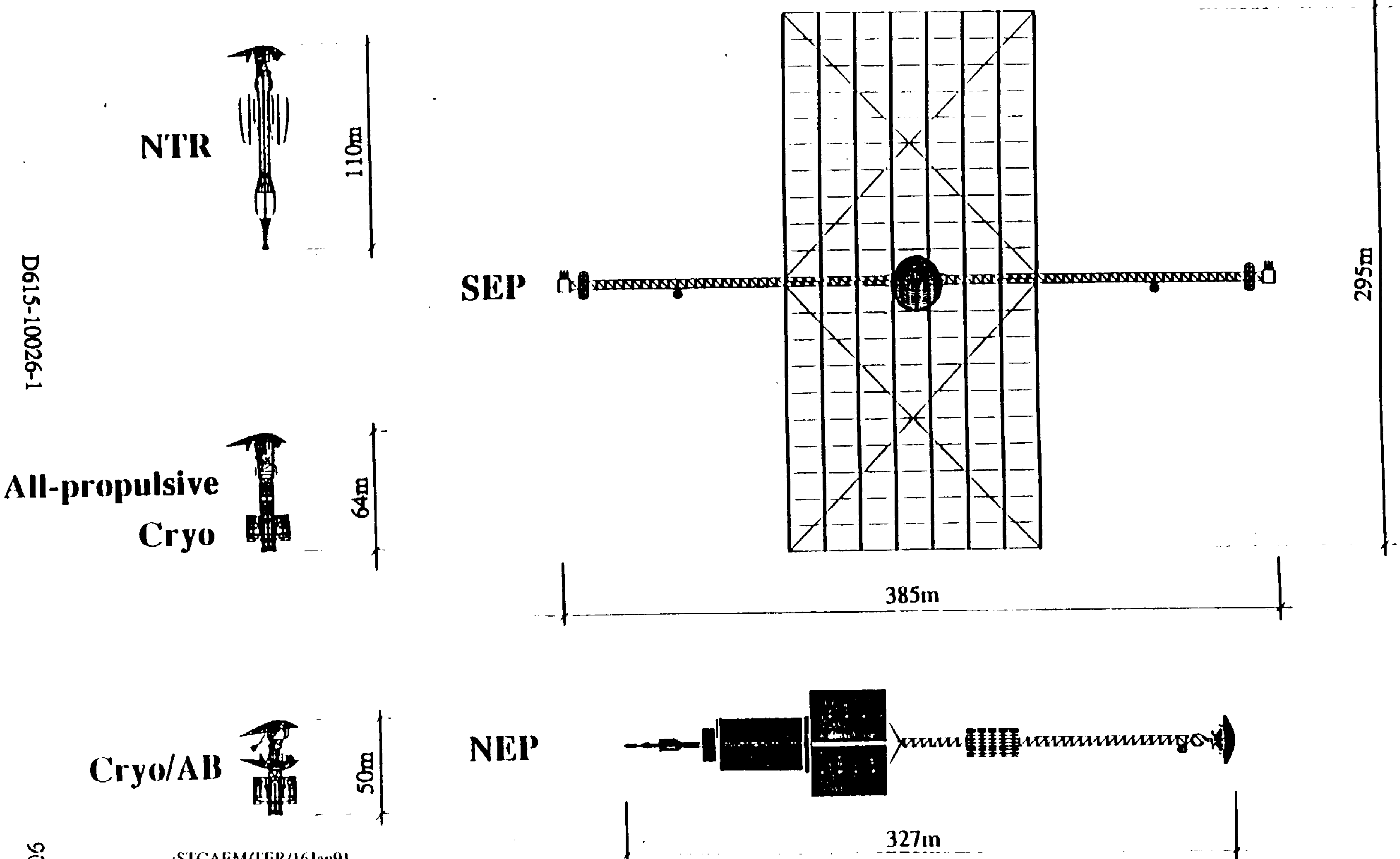
Reference Vehicles

These vehicles, shown on the following chart, represent the configurations as of the third quarter briefing and updates up to Aug. 15, 1990. Manifesting and assembly operations analysis were conducted for the NTR, NEP, and SEP vehicles. The Cryo/Aerobrake operations analyses were presented in the 2nd and 3rd quarter briefings.

Reference Vehicles
Size Comparison



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On-Orbit Assembly Concepts

The following two charts illustrate in rough sketches, assembly platform concepts developed for the Mars vehicles. These concepts range from a totally self-contained platform, such as the ET-based concept (presented in the 2nd quarter briefing), initially developed for the Cryo/Aerobrake vehicle, and the Dedicated Platform concept, initially developed for the NTR, to the vehicle structure itself serving as the assembly platform for NTR, NEP, and SEP vehicles. "Hybrid" concepts, that utilized the vehicle structure with a minimal extraneous structure in a symbiotic mode of operation, were also developed. A case in point is the I-Beam concept that attaches to the NTR or NEP vehicle's truss and rolls along the truss during vehicle assembly. It is emphasized that no platform concept satisfies every vehicle entirely. Therefore, the type of on-orbit assembly platform concept chosen is dependent on the vehicle that is chosen for the Mars mission.

ADVANCED CIVIL SPACE SYSTEMS

On-orbit Assembly Concepts

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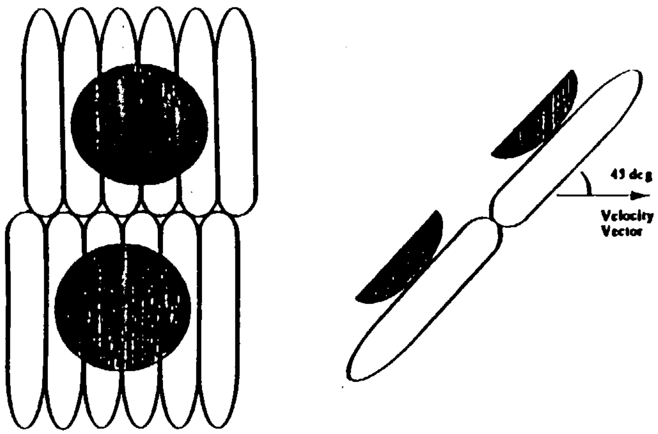
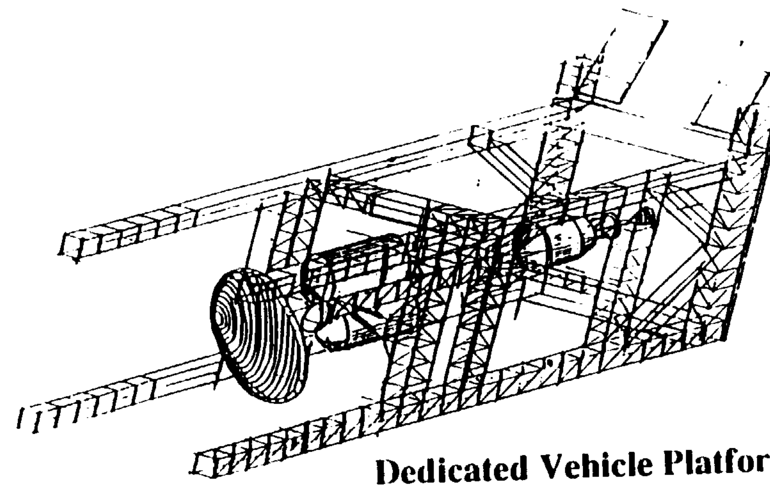
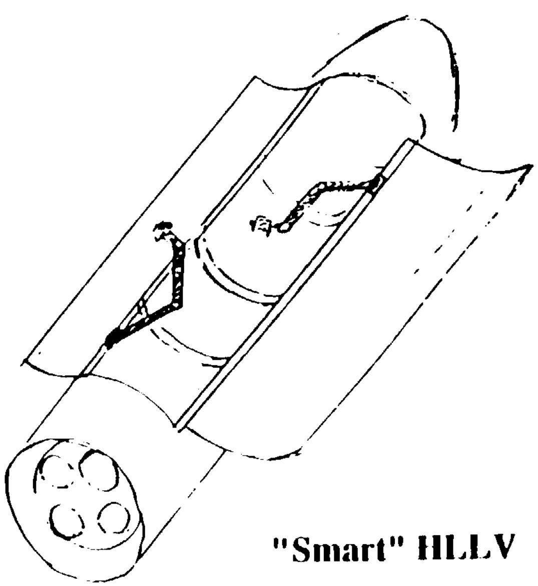


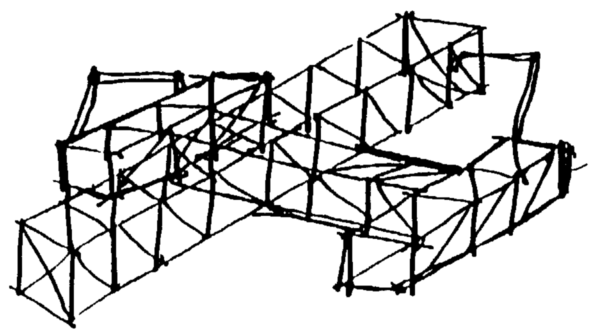
ET-based Platform



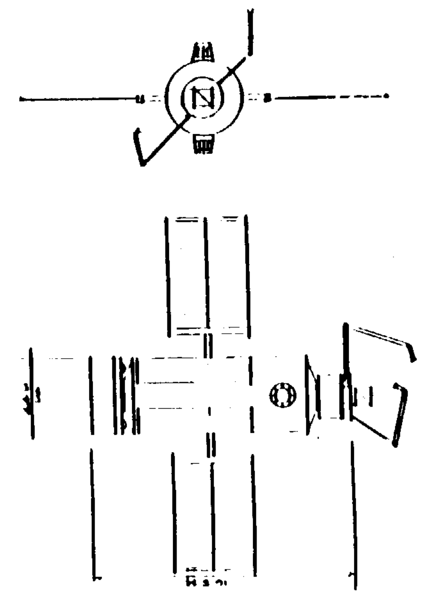
Dedicated Vehicle Platform



"Smart" HLLV



"I-Beam"



Assembly Flyer

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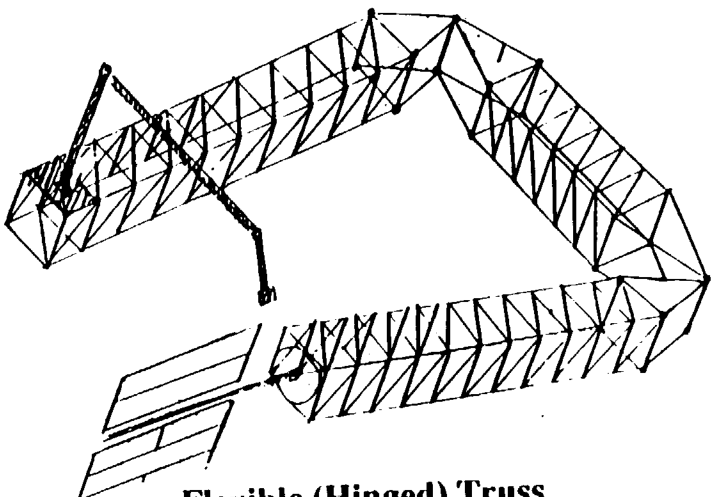
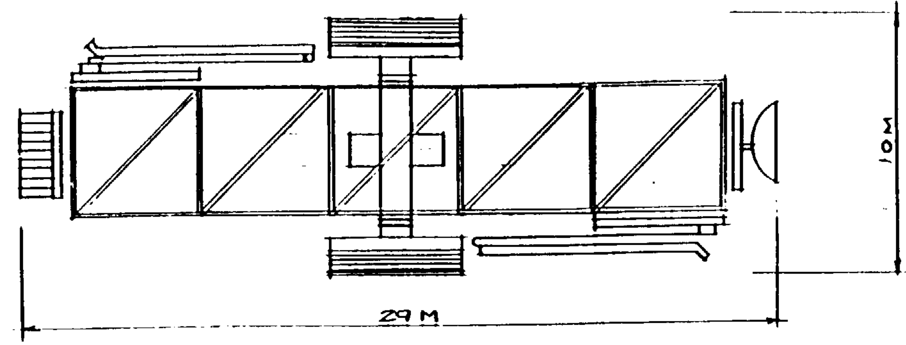
On-orbit Assembly Concepts- continued

Shown on the following chart are two concepts which would utilize Space Station Freedom for assembly of the Mars vehicles. First, planned growth of SSF would accommodate construction of First Element Launch (FEL) of NTR, NEP, or SEP. Once this element is assembled and outfitted appropriately, it could be transported away from SSF and the remainder of Mars vehicle assembly would be based on the vehicle itself. Second, a tether system attached to SSF would allow construction of the Mars vehicle at one "end" and on-orbit propellant depot operations at the other. Both concepts are intended to utilize the existing resources, support, and crew available from SSF without major disruption of SSF operations. The first accomplishes this by only requiring SSF services for one or two assembly missions; the second uses the tethers to optimally locate the center of gravity and to mitigate assembly impacts to SSF.

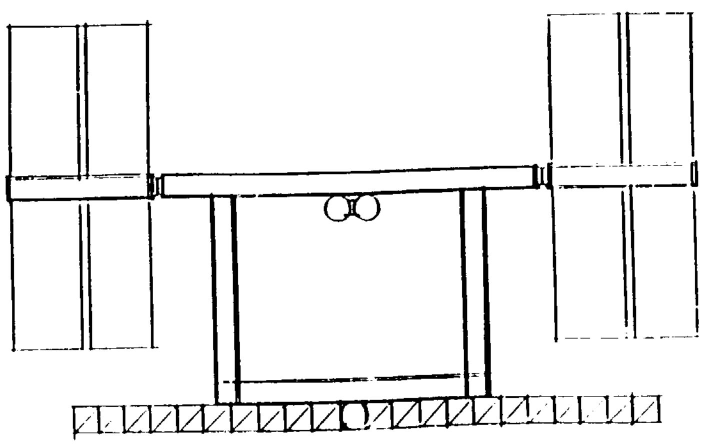
On-orbit Assembly Concepts - continued



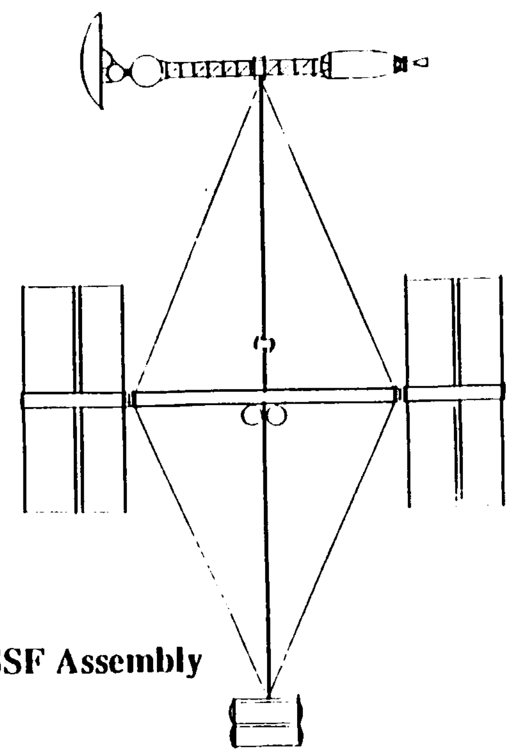
Flexible (Hinged) Truss



Vehicle as Its Own Platform



SSF-based FEL Assembly



Tethered-Off-SSF Assembly

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Assembly Node Concepts Pros and Cons

The following two charts compare each on-orbit assembly platform concept with respects to its key features and deficiencies

Assembly Node Concepts Pros and Cons

Node Concepts	Key Features/Advantages	Key Disadvantages
Dedicated Assembly Node	• Abundant storage • Totally self-contained • Vehicle systems unused • Multiple robot arms • Sections of vehicle may be assembled simultaneously	• Larger than SSF • Will take long time to construct • Excessive reboost requirements • Mechanically complex • Local debris shielding required • Must be in place prior to vehicle assembly
I-Beam Platform	• Can be carried up in first HLLVflight • Can easily reach most parts of vehicle with two robot arms • Uses vehicle for comm., data, RCS, power after initial deployment • Can serve as base for experiments	• Fuel cells, batteries required for initial deployment • Limited storage area • Precursor mission required for deployment
"Smart" HLLV Platform	• No additional platform required • HLLV shroud provides limited debris shielding • HLLV provides for communication, data, RCS, GNC, etc. • Robot arms transferable to NTR	• Increased HLLV complexity • Reboost fuel has to be replenished • Limited storage • Vehicle must be detached from HLLV prior to assembly complete • Local debris shielding required
Hinged Truss Platform	• Uses vehicle truss as assembly platform; no other platform needed • Reach to remote engine section of vehicle provided by flexing truss at hinges • Vehicle subsystems used; no additional systems necessary	• Requires a precursor mission to deploy truss • Batteries, fuel cells necessary for initial deployment • Reboost, comm., data, power, must be in place prior to assembly start • Limited storage • Local debris shielding required
Vehicle as its own Platform	• Reduces needed on-orbit infrastructure • Deletes additional facilities and resources needec for designing, builiding, launching, and maintaining separate assembly platform	• Requires dedicated HLLV flight for non-optimized packaged first element • Requires vehicle to have additional control, reboost • No additional storage • Requires batteries or fuel cells for initial deploymen • Requires localized debris shielding

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Assembly Node Concepts Pros and Cons
(continued)

Node Concepts	Key Features/Advantages	Key Disadvantages
Assembly Flyer Platform	• Performs HLLV unloading, payload/crew transport, and assembly with one vehicle • Compatible with SSF • Capable of manned/robotic operations • Uses CTV for main P/A • Can serve as free flying platform between assemblies	• No additional storage • Requires vehicle to have additional control and reboost systems • Requires development and production of sophisticated man-rated space vehicle • Requires localized debris shielding
SSF Based Assembly of First Element	• Uses planned SSF growth concept • Provides quick and easy crew logistics access to initial assembly operations • Allows verification and checkout of critical systems prior to independent vehicle operations • Does not disrupt SSF operations beyond first assembly mission (remainder of assembly based from vehicle itself)	• Impact to SSF (resources, microgravity, drag, etc.) • Eventually requires vehicle to have additional control and reboost systems • Requires localized debris shielding • No additional storage beyond first element
Tethered off-SSF Assembly Platform	• Compatible with current SSF design • Provides quick and easy crew and logistics access to entire assembly and propellant transfer operations • Microgravity and dynamic loads impacts to SSF minimized by tether • Removes hazardous operations and materials to SSF standoff distance	• Impact to SSF resources • Requires localized debris shielding • No additional storage • Requires additional reboost and control systems on SSF

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On-Orbit Assembly Concepts Summary

The following matrix summarizes the applicability of each assembly concept to each vehicle. 'X's indicate concepts that are applicable to particular vehicles, whereas '?'s indicate that they may or may not be applicable.

On-Orbit Assembly Concepts Summary

On-orbit Assembly Concept	Vehicle Applicability				
	CAB	CAP	NTR	NEP	SEP
• Vehicle as Its Own Platform	----	----	X	X	X
• ET-based Platform	X	X	MEV only	MEV only	MEV only
• Dedicated Vehicle Platform	X	X	X	?	----
• "I-Beam"	----	----	X	X	?
• "Smart" HLLV	FEL	FEL	FEL	FEL	FEL
• Flexible (Hinged) Truss	----	----	?	X	?
• Assembly Flyer	?	?	X	X	X
• SSF-based FEL Assembly	FEL, MEV (MTV)	FEL, MEV	FEL, MEV	FEL, MEV	FEL, MEV
• Tethered-Off-SSF Assembly	X	X	X	----	----

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NTR Component Manifest Data

These data represent the vehicle configurations as of the third quarter briefing. Similar data exist for the NEP and SEP vehicles as well but are not included in this briefing.

ADVANCED CIVIL SPACE SYSTEMS

NTR Component Manifest Data

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NTR Component	Quantity	Dimensions (meters)	Total Mass (metric tons)
• MEV			
Aeroshell	1	28 x 30 x 7 box	9.51
Descent System (incl 2 rovers)	1	9.5 x 20 x 4 box *	32.83
Ascent System	1	9.5 x 9.5 x 5.5 box	24.83
Surface Payload Module	1	13 x 4.4 (dia) cylinder	25.00
Surface Payload Module Airlock	1	2.9 x 3 (dia) cylinder	4.50
		Subtotal =	**96.67**
• MTV			
MTV Hab Module	1	10 x 8 (dia) cylinder	40.30
MCRV	1	3 x 4 (dia) cylinder	7.00
MTV-to-MEV Tunnel and Airlock	1	6 x 3 (dia) cylinder	7.00
Main Truss (includes RCS, RCS fuel, main fuel lines, mass growth, GNC)	1	7 x 7 x 7 box (deployable) *	5.6
Mars Orbit Capture Tanks (include fuel)	2	20 x 10 (dia) cylinders	178
Trans-Mars Injection Tanks (include fuel)	2	30 x 10 (dia) cylinders	329
In-Line Tank (includes fuel)	1	19 x 10 (dia) cylinder	101
Tank-to-Truss Structure	1	11.5 x 3	1.00

* These represent launch package dimensions, not mission configuration

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NTR Component Manifest Data- continued

NTR Component	Quantity	Dimensions (meters)	Total Mass (metric tons)
• MTV (continued)			
Tank-to-Shield Structure	1	7 x 7	2.40
Shield/Engine	1	10 x 7 (dia)	11.80
		Subtotal =	683.10
		NTR Total =	779.77

NTR - Manifesting and Packaging

Schematics below depict basic packaging of NTR components for each assembly flight. This manifest utilizes the single-class HLLV fleet and "smart" HLLV platform concept. Protrusions into the nosecone, where shown, are assumed to be workable. Not accounted for here are FSE/OSE required or any assembly support/resupply equipment. Integrated aerobrake concept (ninja turtle) could possibly save two flights.

NTR - Manifesting and Packaging

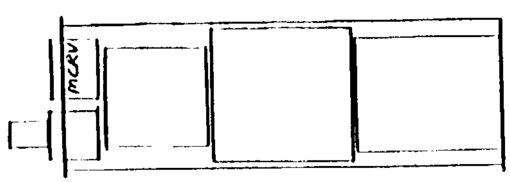
(10m x 30m Shroud, 140 mt HLLV using "Smart" HLLV Platform)

Ground Rules and Assumptions

- Heavy Lift Launch Vehicle (HLLV) with 140 metric ton capability and 10m x 30m shroud
- Sequencing based on "smart" HLLV Assembly Platform concept
- Only first HLLV is "smart"; shroud is not shed in order to provide partial debris protection
- "Smart" HLLV contains all necessary data, communications, rcs, reboost equipment; occasional refueling may be provided by SSF based CTV
- Some TBD volume is available in nosecone of HLLV
- No specific FSE/OSE or CG constraints identified (heavier payload located at bottom of stack)
- NTR configuration (volume and mass) current as of 3rd Quarterly Briefing
- MEV Aeroshell is assembled on orbit (in ten pieces) and requires two dedicated HLLV flights
- 50 mt of propellant off-loaded from TMI tanks, are carried up in tankers along with MOC tanks missions four and five

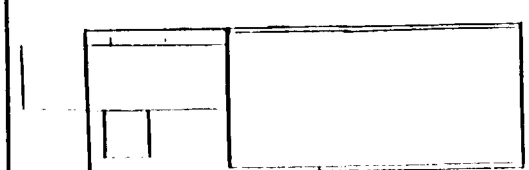
Assembly Mission One

- NTR Truss
- MTV Hab Module
- MEV Ascent stage
- MTV-to-MEV Airlock and Tunnel
- MCRV



Assembly Mission Two

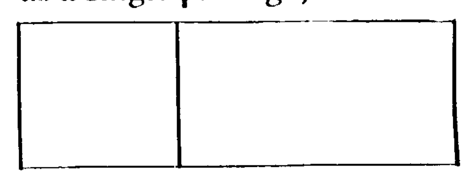
- MEV Descent Stage
- Surface Payload Module
- Airlock for Surface Payload Module (integrated with surface module)



Assembly Mission Three

- In-Line Tank + Fuel
- Tank-to-Truss Structure
- Tank-to-Shield Structure
- Engine/Shield

(these components are integrated as a single package)



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NTR - Manifesting and Packaging (continued)

(10m x 30m Shroud, 140 mt HLLV using "Smart" HLLV Platform)

Assembly Mission Four

- Mars Orbit Capture Tank and fuel (1 of 2)
- Off-loaded fuel tanker (1 of 2)

Assembly Mission Five

- Mars Orbit Capture Tank and fuel (2 of 2)
- Off-loaded fuel tanker (2 of 2)

Assembly Mission Six

- TMI Tank + Fuel (1 of 2)

Assembly Mission Seven

- TMI Tank + Fuel (2 of 2)

Assembly Mission Eight

- MEV Aeroshell (5 Sections)

Assembly Mission Nine

- MEV Aeroshell (5 Sections)

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NTR- Manifesting and Packaging

In this manifest of the NTR vehicle, a mixed HLLV fleet is utilized. Results showed that this offered no advantage to the assembly of this vehicle. In fact, carrying up the aerobrake disassembled, would add at least two flights over the single-class HLLV fleet.

NTR - Manifesting and Packaging

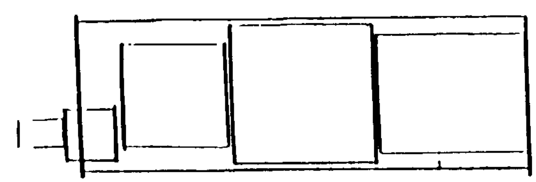
(Mixed HLLV Fleet, using "Smart" HLLV Platform)

Ground Rules and Assumptions

- Heavy Lift Launch Vehicle (HLLV) mixed fleet consists of:
  - HLLV #1: 84 metric ton payload capability with 10m x 30m shroud
  - HLLV #2: 120 metric ton payload capability with 7.6m x 30m shroud
- Sequencing based on "smart" HLLV Assembly Platform concept
- Only first HLLV is "smart"; it contains all necessary communications, data, rcs, GNC equipment
- Occasional refueling of reboost system may be necessary and may be accomplished by SSF based CTV
- Some TBD volume is available in nosecone of HLLV
- No specific FSE/OSE or CG constraints identified (heavier payload located at bottom of stack)
- NTR configuration (volume and mass) current as of 3rd Quarterly Briefing
- MEV Aeroshell is assumed to be integrated at launch ("Ninja Turtle" concept) with other payload packaged in shroud
- Total of 203 tons of propellant, off-loaded from all tanks, are carried up in two tankers in fueling missions eight and nine

Assembly Mission One (HLLV #1)

- NTR Main Truss
- MEV Ascent System
- MTV Hab Module
- MTV-to-MEV Airlock and Tunnel



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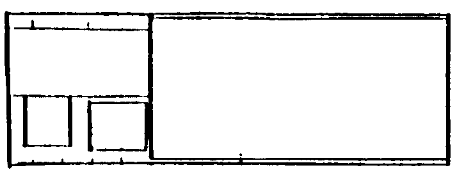
NTR - Manifesting and Packaging (continued)

(Mixed HLLV Fleet, using "Smart" HLLV Platform)

Assembly Mission Two (HLLV #1)

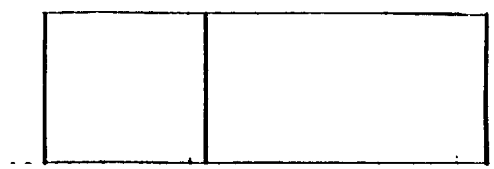
- MEV Descent Stage (includes 2 rovers)
- Mars Surface Payload Module
- Mars Surface Module Airlock (integrated with surface module)
- MCRV
- Aerobrake



Assembly Mission Three (HLLV #1)

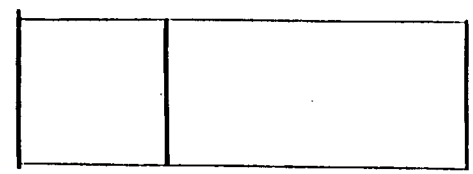
- In-Line Tank + Fuel
- Tank-to-Truss Structure
- Tank-to-Shield Structure
- Engine/Shield

(These components are integrated as a single package)



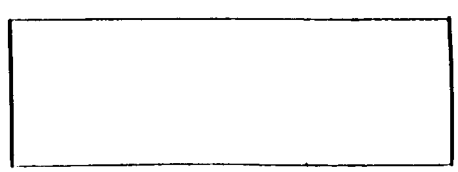
Assembly Mission Four and Five (HLLV #1)

- Mars Orbit Capture Tanks + Fuel (2)



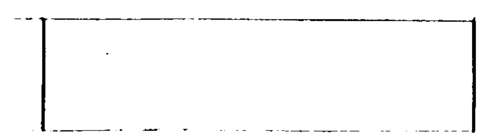
Assembly Mission Six and Seven (HLLV #1)

- TMI Tanks + Fuel (2)



Assembly Mission Eight and Nine (HLLV #2)

- Off-Loaded fuel Tankers and total of 203 tons of fuel



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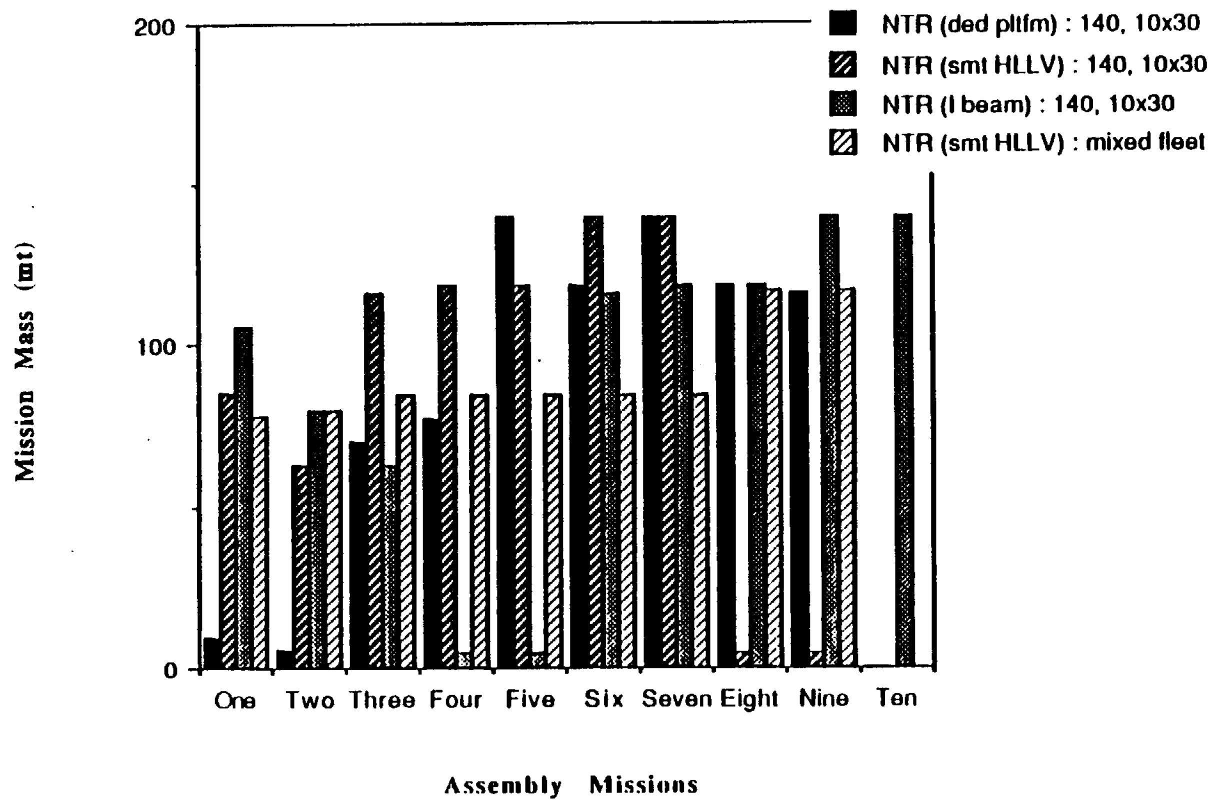
NTR- Manifesting and Packaging

This chart compares the NTR mission mass with respect to four assembly platform concepts. The I-Beam concept manifest consists of 10 missions, whereas the others are nine. The reason for this additional flight is that the I-Beam platform is carried up in the first HLLV flight. The mixed fleet manifest assumes that the aerobrake is launched integrated; launching it disassembled would add two more flights. The last two flights in the mixed fleet are for off-loaded propellant required to fill each tank of the NTR vehicle.

NTR Manifesting and Packaging

Comparison of Manifest Data for NTR Vehicle for Several On-Orbit Platform Concepts



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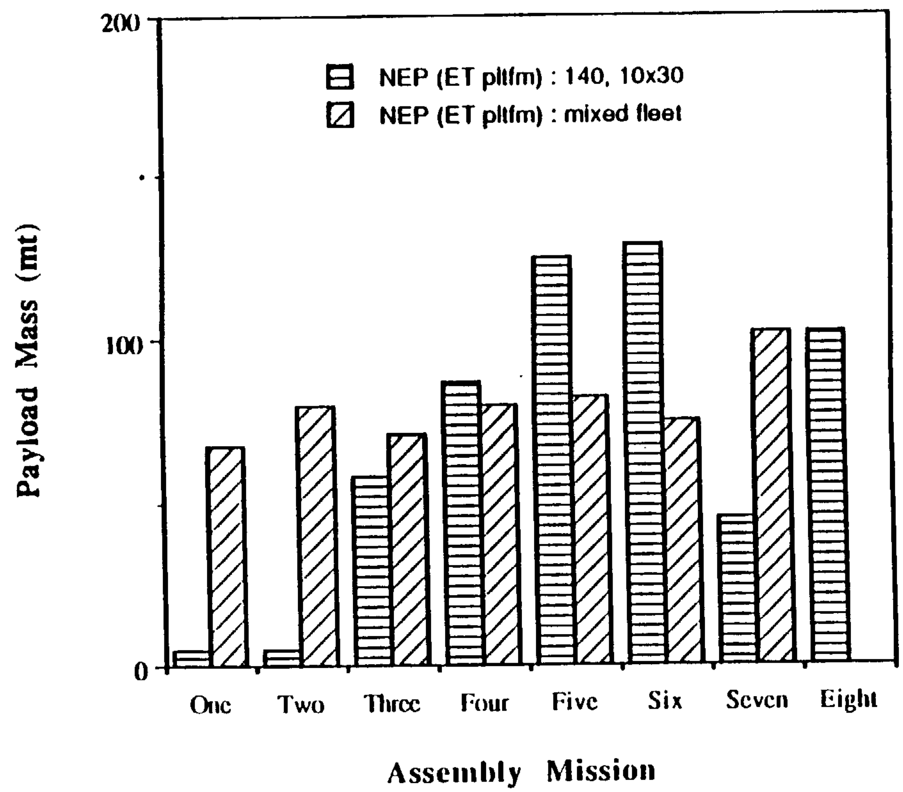
NEP Manifesting and Packaging

Similarly, the NEP was manifested for both a single-class HLLV fleet and a mixed HLLV fleet with respect to the ET-based platform concept. The mixed fleet manifest for the NEP also assumes that the aerobrake is launched integrated; assembling the aerobrake on-orbit would add two more flights for a total of nine.

The advantage of a mixed fleet, even though it may add flights, is that each launch package may better utilize the mass and volume capabilities of a particular HLLV. An integrated aerobrake launch improves packaging efficiency even further.

NEP Manifesting and Packaging

Comparison of Payload Mass per Assembly Mission for NEP Using Different HLLV Fleets



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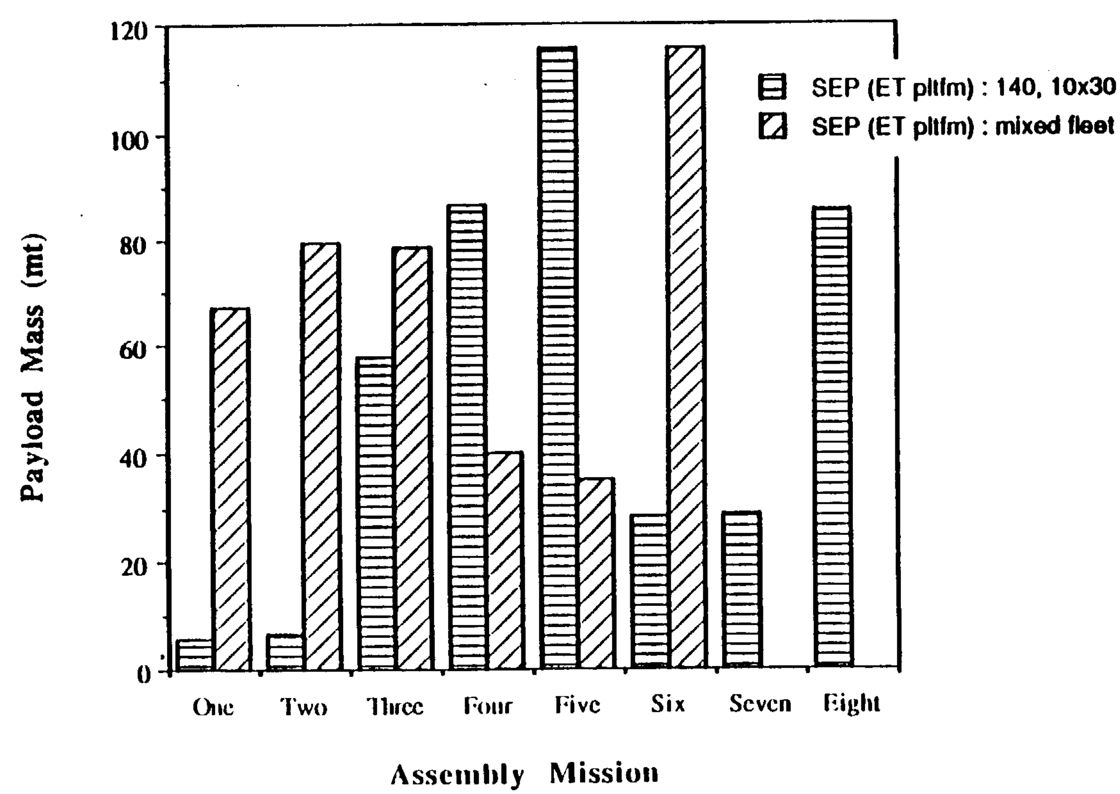
SEP Manifesting and Packaging

Similar results as discussed for NEP.

SEP Manifesting and Packaging

Comparison of Payload Mass per Assembly Mission for SEP Using Different HLLV Fleets



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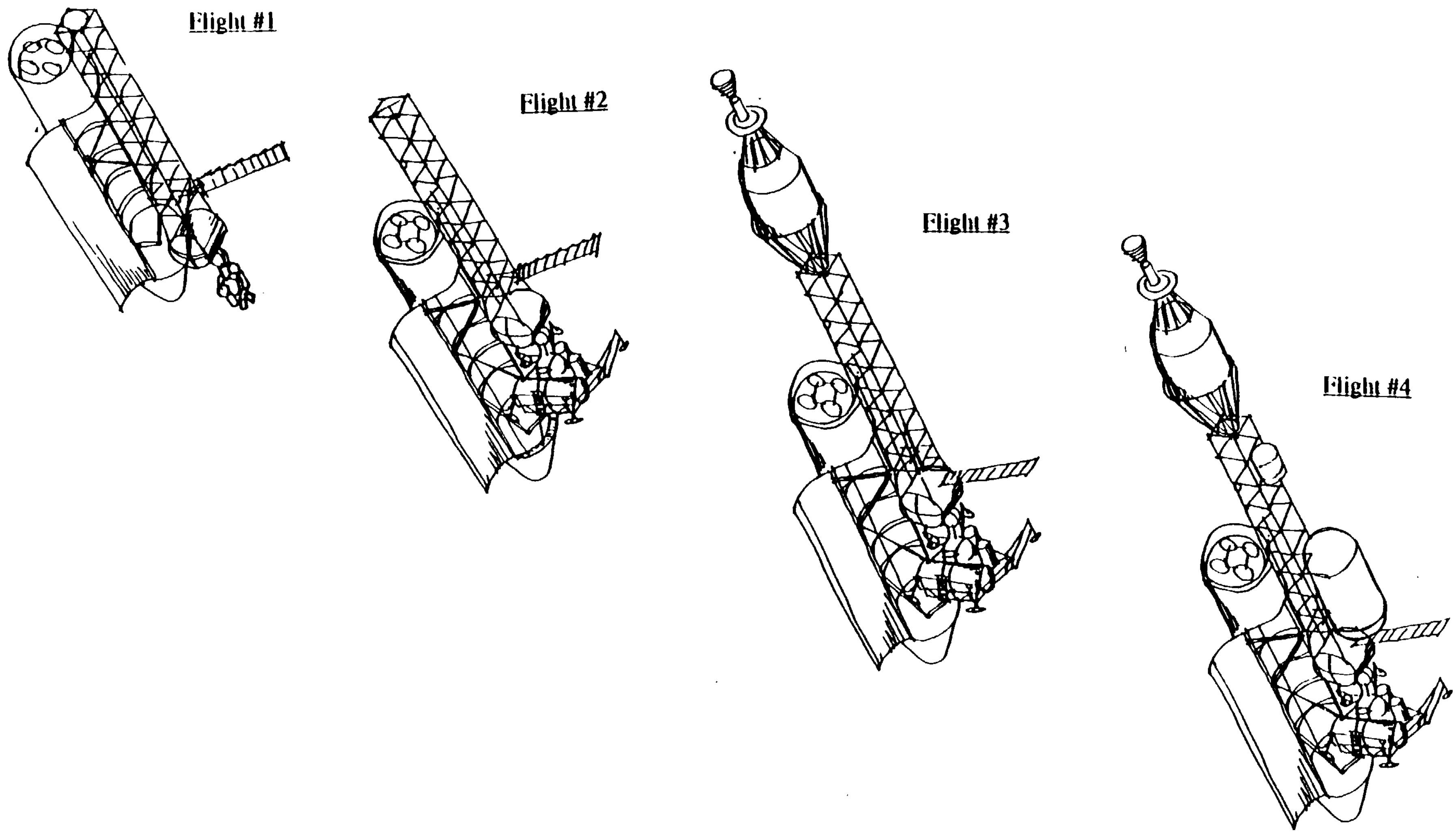
"Smart" HLLV Platform: NTR Assembly

The next two charts illustrate in rough sketches the NTR vehicle in various phases of assembly corresponding to the assembly flights manifest using the "smart" HLLV assembly platform concept and using a single-class HLLV fleet of launch vehicles (not a mixed fleet). Only the first HLLV is "smart", i.e., it is launched to LEO with shroud intact and it contains the necessary smarts (communication, data, reboost, robot arms) to serve as an assembly platform through the fourth mission. Following the fourth mission, the vehicle is detached from the HLLV platform to enable attachment of the remaining propellant tanks, aerobrake. Debris shielding must be locally provided to needed sections of the NTR.

"Smart" HLLV Platform: NTR Assembly



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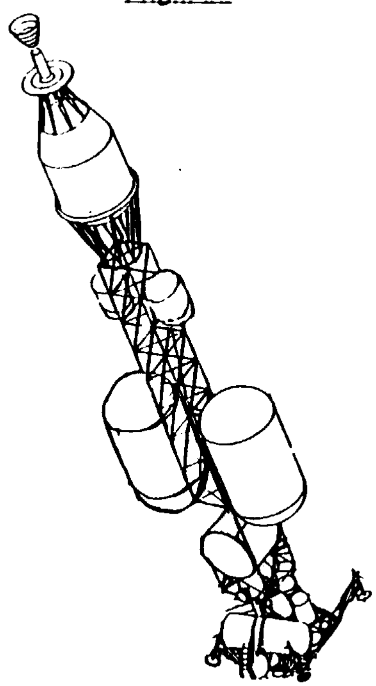
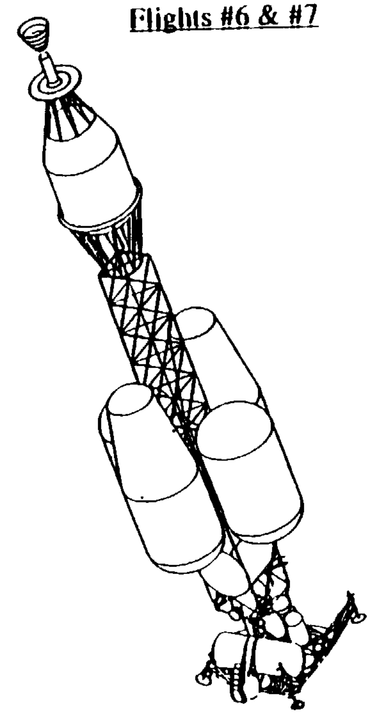
"Smart" HLLV Platform: NTR Assembly

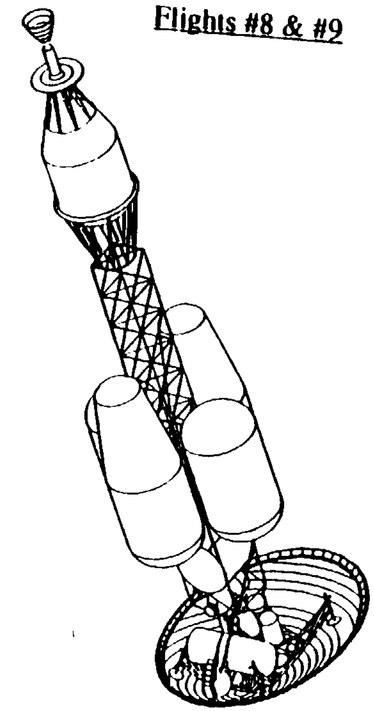
Flight #5



Flights #6 & #7



Flights #8 & #9



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NTR Top Level Assembly Using Dedicated Platform Concept

Detailed on-orbit assembly analyses were performed for the NTR, NEP, and SEP vehicles using MacProject II, similar to what was accomplished for the Cryo/Aerobrake vehicle in the 2nd and 3rd quarters of this study. The assumptions that served as the bases for the NTR, NEP, and SEP assembly flows, were identical to the Cryo/Aerobrake assumptions. The following chart is a top level flow from MacProject, for the NTR vehicle. Detailed flows appear in the Implementation Plan and Element Description (IPED) documents.

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NTR Top Level Assembly Using Dedicated Platform

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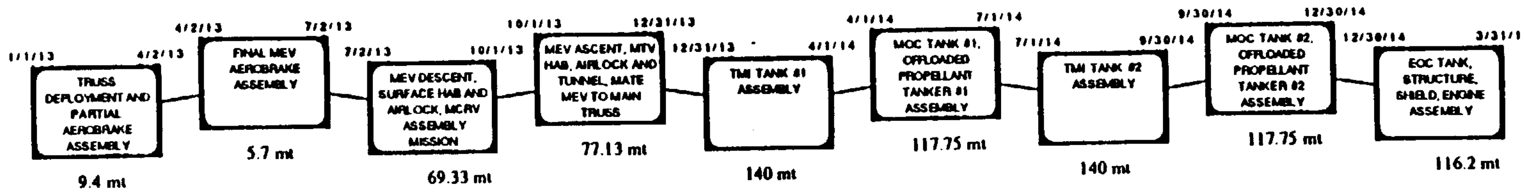

- Smallest unit of time is 1 hour
- 16 hours = 1 day of Assembly Duration

BASELINE DURATIONS:
- HLLV Launch = .5 day
- HLLV achieves stable orbit = .25 day
- MUV deploys from/to Freedom = .5 day
- MUV berths to components = .25 day
- Unstow and power up Robotics = .06 day
- Robotic verification = .12 day
- HLLV deploys components = .06 day
- MUV transfers components = .25 day
- Robotic tasks = .06 day
- EVA/Robotic Contingency = .5 day
- Component Inspection = .12 day
- Component Test = .25 day
- Subassemblies to stand-by mode = .5 day
- Mechanical Fastening of components = .18 day
- Crew processing for flight = .25 day

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NTR Top Level Assembly Schedule

This chart shows the assembly corresponding to the mission manifest. 90-day intervals are assumed between HLLV flights.

NTR Top Level Assembly Schedule

ADVANCED CIVIL SPACE SYSTEMS

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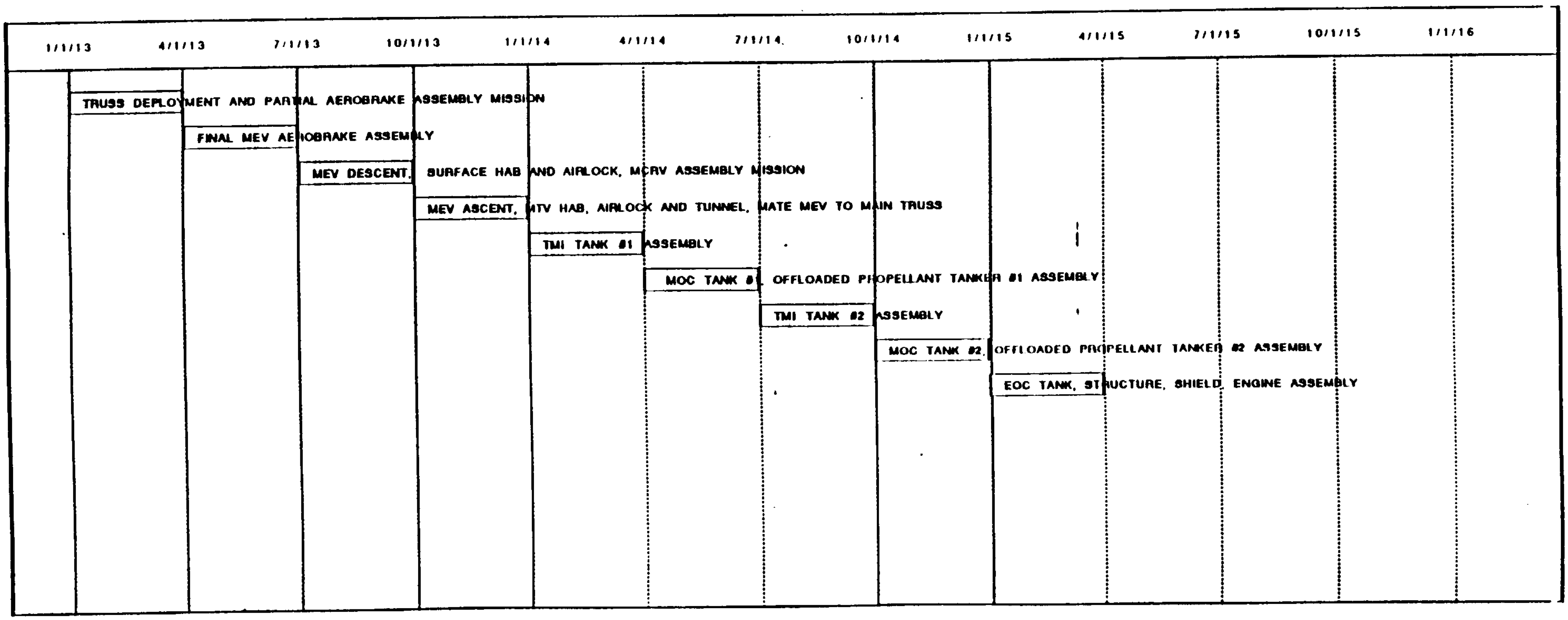
Name	Earliest Start	Earliest Finish	Subproject	Days
TRUSS DEPLOYMENT AND PARTIAL AEROBRAKE ASSEMBLY MISSION	1/1/13	4/2/13	NTR MISSION ONE	91
FINAL MEV AEROBRAKE ASSEMBLY	4/2/13	7/2/13	NTR MISSION TWO	91
MEV DESCENT, SURFACE HAB AND AIRLOCK, MCRV ASSEMBLY MISSION	7/2/13	10/1/13	NTR MISSION THREE	91
MEV ASCENT, MTV HAB, AIRLOCK AND TUNNEL, MATE MEV TO MAIN	10/1/13	12/31/13	NTR MISSION FOUR	91
TMI TANK #1 ASSEMBLY	12/31/13	4/1/14	NTR MISSION FIVE	91
MOC TANK #1, OFFLOADED PROPELLANT TANKER #1 ASSEMBLY	4/1/14	7/1/14	NTR MISSION SIX	91
TMI TANK #2 ASSEMBLY	7/1/14	9/30/14	NTR MISSION SEVEN	91
MOC TANK #2, OFFLOADED PROPELLANT TANKER #2 ASSEMBLY	9/30/14	12/30/14	NTR MISSION EIGHT	91
EOC TANK, STRUCTURE, SHIELD, ENGINE ASSEMBLY	12/30/14	3/31/15	NTR MISSION NINE	91



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Ground Rules and Assumptions for Ground Processing

Groundrules and assumptions used for the NTR, NEP, and SEP ground processing are similar to those used for the Cryo/Aerobrake vehicle ground processing analyses. One noteworthy difference is the last assumption which states that certain electrical and fluid interfaces to the NTR, NEP, and SEP will be simulated if launch sequencing can be improved. For instance, this provision would allow launching the MTV hab earlier in the assembly sequence requiring the rest of the interfaces to the MTV hab to be simulated.

Ground Rules and Assumptions for Ground Processing

- A system is a group of components and supporting structure that is integrated by a contractor and delivered as a unit to the processing facility (e.g. MEV Aerobrake, MEV descent lander, ascent system, etc.).
- System interfaces are those which transmit data, power, or fluids across the system's boundaries and mechanically secure one system to another.
- Subsystems interfaces are those which are internal to a system.
- Subsystem interfaces are verified by the manufacturer prior to system integration.
- Component interfaces are those which are internal to a subsystem.
- Component interfaces are verified by the manufacturer during subsystem assembly.
- Interfaces verified prior to a system level integration will be accepted with no repetition of tests.
- Flight hardware will be used to verify system interfaces.
- Ground facilities will simulate assembly node operations and limitations.
- Certain non-mechanical interfaces to NTR, NEP, and SEP are simulated to allow desired launch sequencing.

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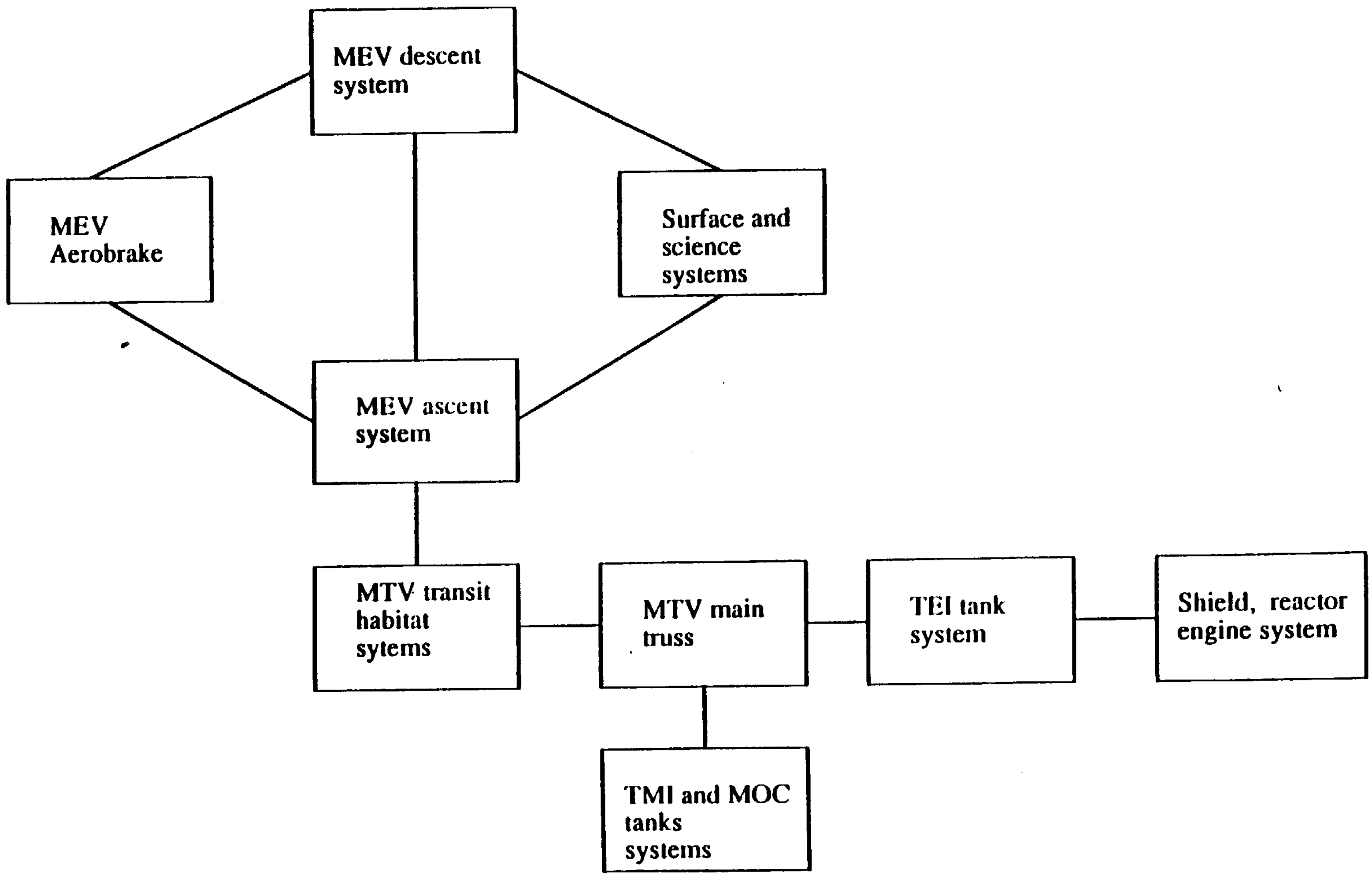
NTR System Interfaces (Top Level)

Interface analysis was performed for the NTR, NEP, and SEP vehicles again similar to what was accomplished for the Cryo/aerobrake vehicle. This chart is a sample of a top level interface diagram for the NTR vehicle.

NTR MMV System Interfaces (Top Level)



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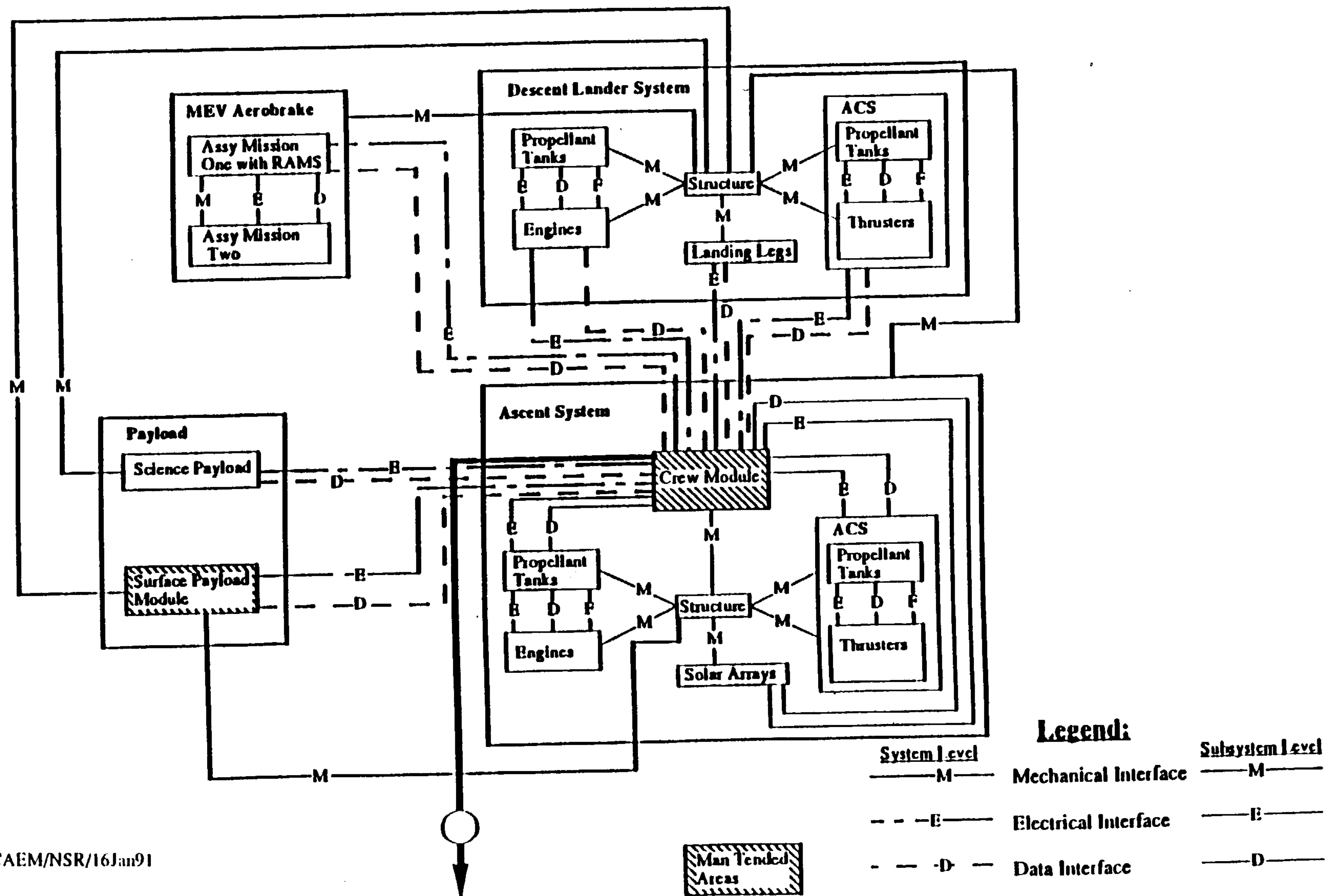
NTR System Interfaces (MEV)

The next two charts show the major mechanical, fluid, electrical, and data interfaces and pathways associated with the MEV and MTV systems repectively, at a more detailed level.

NTR System Interfaces

MEV System Interfaces



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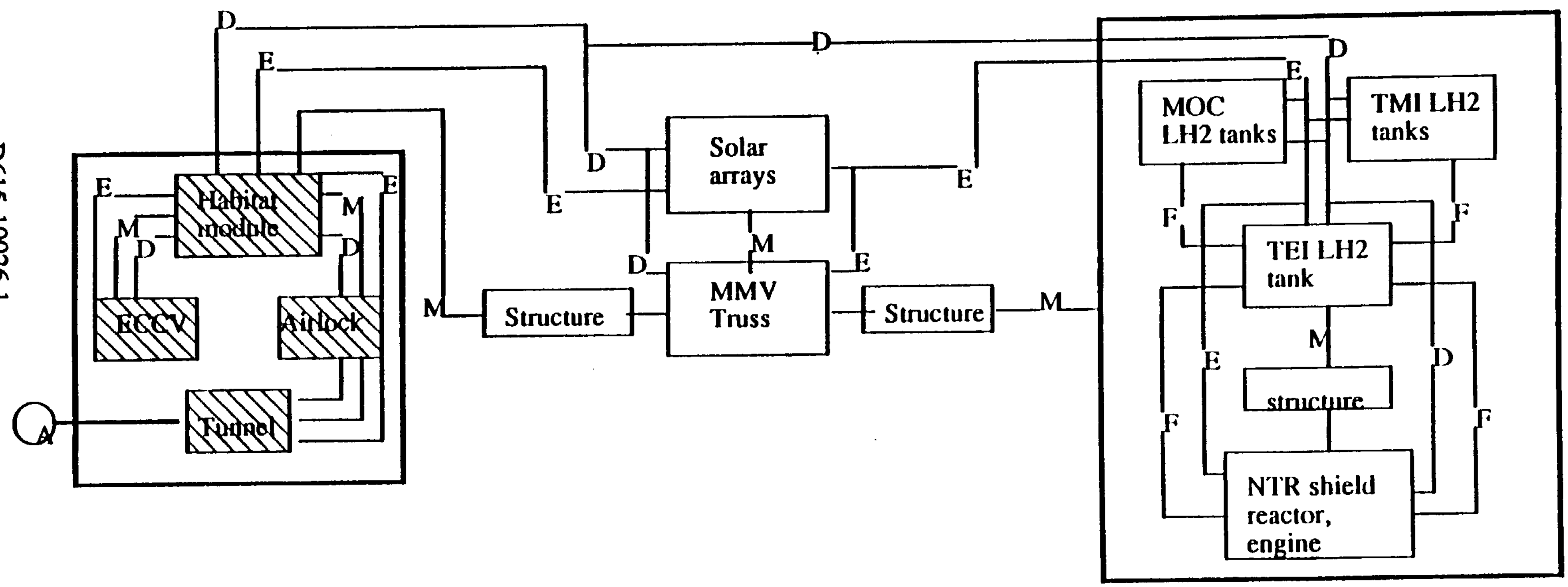
MTV System Interfaces



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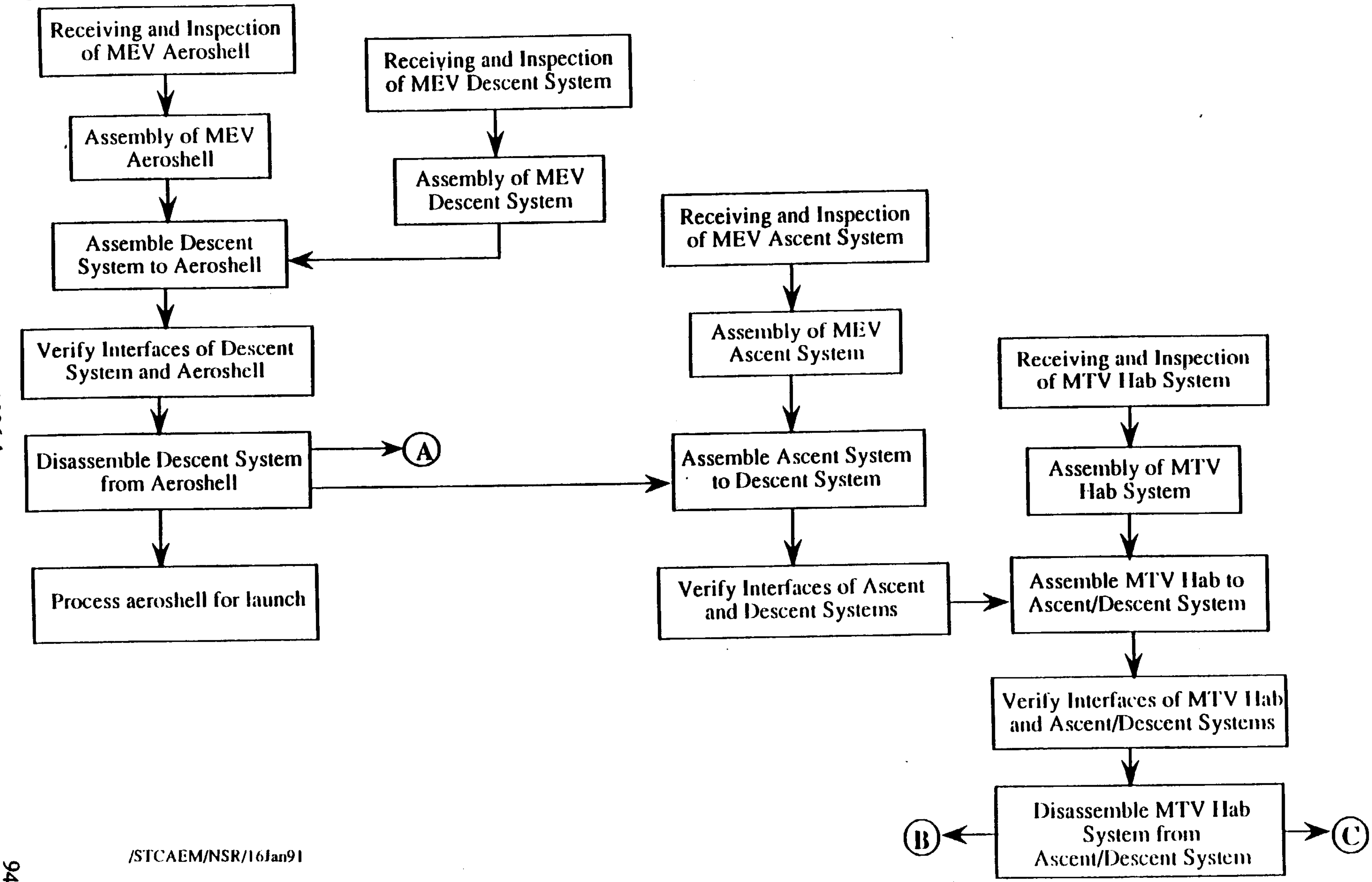
NTR Ground Processing Functional Flows

High level functional flows for ground processing for launch of the NTR, NEP, and SEP vehicles components were developed. The following three charts show the NTR ground processing flow. Detailed analyses of scheduling ground processing with on-orbit assembly must be performed as these are heavily interdependent.

NTR Ground Processing Functional Flow



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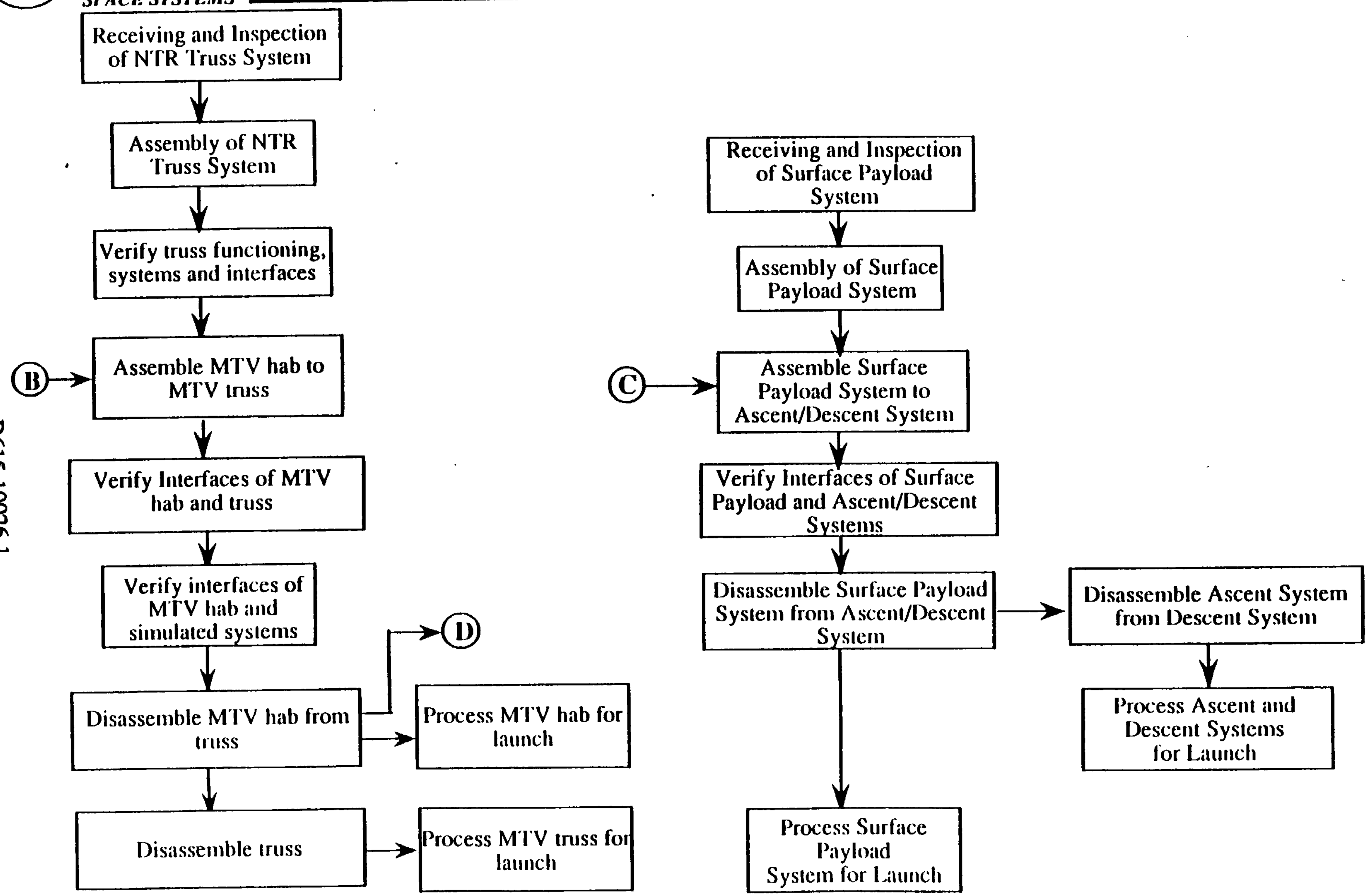
NTR Ground Processing Functional Flow - continued



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NTR Ground Processing Functional Flow - continued

ADVANCED CIVIL SPACE SYSTEMS

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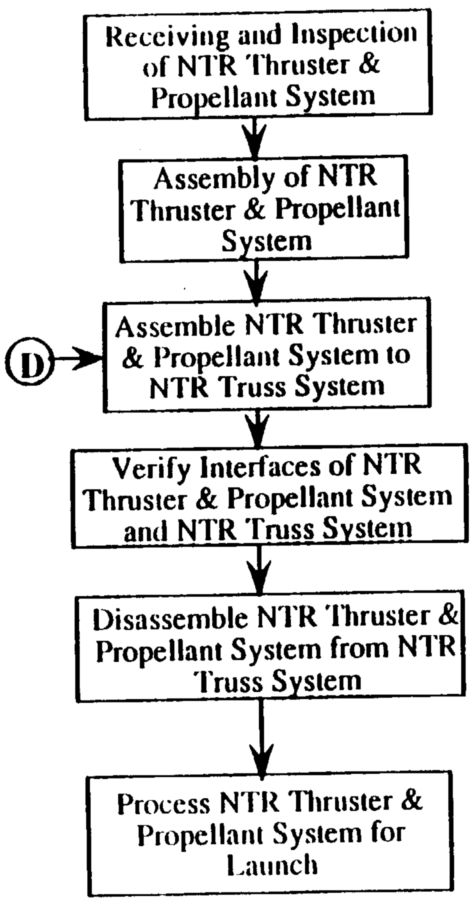

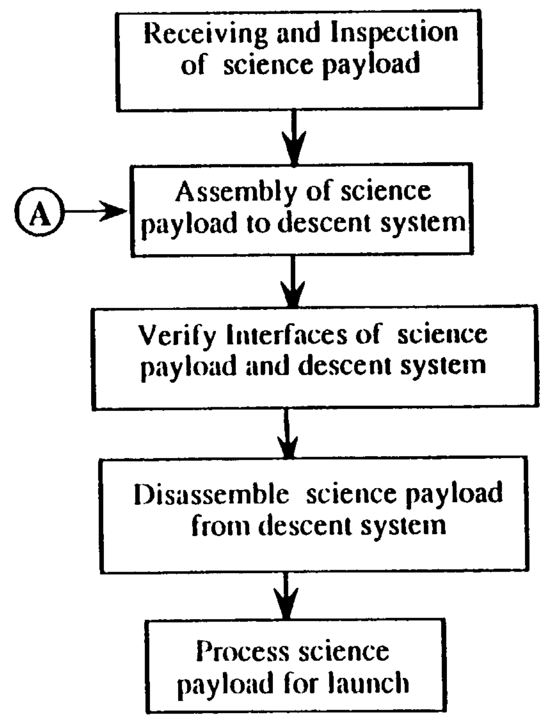

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Special Ground and On-Orbit Processing Facility and Equipment Requirements

This chart outlines the special facilities and equipment that would be required for each of the three vehicles. This list is in addition to the list that was generated for the Cryo/Aerobrake vehicle in the 3rd quarter briefing. Volume requirements analyses for ground facilities were not conducted but will be associated with future mission manifests and ground processing.

Special Ground and On-Orbit Processing Facility and Equipment Requirements

Facilities/Equipment	NTR	NEP	SEP
Ground			
• Reactor/engine mating and processing facility	X	X	
• Nuclear fuel loading facility	X	X	
• Contaminated materials storage and disposal facility	X	X	
• Solar array/radiator packing and storage facility	X	X	X
• Alkali metals materials and transferring facility		X	
• Radiation/hazardous materials contamination treatment facility	X	X	
• Robotics to handle radioactive fuels and hazardous chemicals/materials and components	X	X	
• Vehicle truss processing and packaging facility	X	X	X
On-Orbit			
• On-orbit robotic welding and certification equipment	X	X	
• On-orbit alkali metal heating capability		X	
• On-orbit robotic repair/maintenance equipment	X	X	X

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Summary (Ground Processing, Manifesting, On-Orbit Assembly)

- Ground processing flows are very interdependent upon the launch vehicle and assembly concept assumed
- Non-hardware interface verification may require simulators to better schedule hardware deliveries
- Assembly flights are mainly volume, not mass, dependent
  - Most flights underutilize relative mass capability
  - A mixed fleet may improve launch packaging efficiency for NEP and SEP
  - Integrated aerobrake launch provides advantage in terms of number of flight and orbital assembly
- Capabilities, requirements of first element launch (FEL) of Mars vehicles drives on-orbit assembly infrastructure
- Two of the NEP assembly stages require nearly the full 90 days allotted between flights
  - Radiators and heat transport system require a large number of operations
  - Changes in assumptions used for number of pieces and method of attachment could easily violate 90 day limit
- Assumed deployable truss for NEP, SEP, and NTR reduces on-orbit times
- Assumed extensive assembly robotics tends to decrease crew time and needed infrastructure

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Analysis of Orbital Propellant Depots for the Space Transfer Concepts and Analysis for Exploration Missions Study

Preliminary Study Final Report

December 1990

GENERAL DYNAMICS
Space Systems Division

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Analysis of Orbital Propellant Depots for the Space Transfer Concepts and Analysis for Exploration Missions Study

Preliminary Study Final Report

December 1990

Prepared for
Boeing Aerospace
P.O. Box 1470
Huntsville, Alabama 35807

Prepared under
Contract HG1420

Prepared by
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San Diego, California 92138

FOREWORD

This report was prepared by General Dynamics Space Sytems Division (GDSS) for Boeing Aerospace under Contract HG1420. This report documents results of a seven month technical effort carried out from June 1990 through December 1990.

The Boeing Aerospace technical manager was Gordon Woodcock.

The following GDSS personnel contributed to the report:

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Scott C. Honkonen	Thermodynamics
Nathan A. Nottke	Operations Analysis
Luis R. Pena	Systems Engineering
Matthew W. Thompson	Design
Peter S. Wreschinsky	Computer Mockup

This following cross reference identifies the section number of the report that corresponds to the contract Statement of Work (SOW) task.

Report Section Number	SOW Task
2	5.1.3 (Task 1)
3	5.3.4 (Task 3)
4	5.4.3 (Task 4)
5	5.5.2 (Task 5)
6	5.7 (Task 7)
7	5.8

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LIST OF CHARTS

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ACRONYM LIST

ASTV	Advanced Space Transfer Vehicle
CAP	Cryogenic All Propulsive
CER	Cost Estimating Relationship
CONE	Cryogenic On-orbit Nitrogen Experiment
CY	Calendar Year
DDT&E	Design Development Test & Evaluation
EOC	Earth Orbit Capture
ET	External Tank
ETO	Earth To Orbit
EVA	Extra Vehicular Activity
GaAs	Gallium Arsenide
GEO	Geosynchronus Earth Orbit
GN&C	Guidance Navigation & Control
HLLV	Heavy Lift Launch Vehicle
HMO	High Mars Orbit
IOC	Initial Operational Capability
IVA	Intra Vehicular Activity
LAD	Liquid Acquisition Device
LCC	Life Cycle Cost
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LLO	Low Lunar Orbit
LMO	Low Mars Orbit
LN2	Liquid Nitrogen
LOX	Liquid Oxygen
LO2	Liquid Oxygen
LTCSF	Long Term Cryogenic Storage Facility
LTV	Lunar Transfer Vehicle
MEV	Mars Excursion Vehicle
MLI	Multi-Layer Insulation
MOC	Mars Orbit Capture
MSFC	Marshall Space Flight Center
MTS	Mars Transfer System
MTV	Mars Transfer Vehicle

ACRONYM LIST (Cont.)

NASA	National Aeronautics and Space Administration
NTR	Nuclear Thermal Rocket
R&D	Research and Development
RCS	Reaction Control System
ROM	Rough Order of Magnitude
RMS	Remote Manipulator System
SSF	Space Station Freedom
STCAEM	Space Transfer Concepts & Analysis for Exploration Missions
STS	Space Transportation System
T	Temperature
TDRSS	Tracking & Data Relay Satellite system
TEI	Trans-Earth Injection
TMIS	Trans-Mars Injection Stage
TVS	Thermodynamic Vent System
VCS	Vapor Cooled Shield
WBS	Work Breakdown Structure

SECTION 1

INTRODUCTION

The United States is entering an era of expanded space exploration activity that will involve manned missions to Mars. NASA Marshall Space Flight Center (NASA/MSFC) has thus funded a study of Space Transfer Concepts and Analysis for Exploration Missions (STCAEM). A key aspect of this study is the utilization of high energy, cryogenic propellant for both Chemical Propulsive and Thermal Nuclear Rocket vehicles. Boeing Aerospace has contracted General Dynamics Space Systems Division to analyze orbital propellant depots for these mission scenarios and vehicle concepts.

The objectives of the orbital propellant depot analysis were to: (1) review requirements for orbital depots, (2) perform preliminary trades for orbital depot, location, configuration, and operation, (3) perform analysis of integration compatibility of mission vehicles and depots, (4) develop initial depot concepts, (5) identify depot technology requirements, and (6) estimate non-recurring and recurring depot facility costs. These objectives are summarized in the chart "Orbital Propellant Depot Study Objectives". This report summarizes the initial findings of this preliminary study.

SECTION 1

INTRODUCTION

2

Orbital Propellant Depot Study Objectives

The objectives of the orbital propellant depot analysis were to: (1) review requirements for orbital depots, (2) perform preliminary trades for orbital depot, location, configuration, and operation, (3) perform analysis of integration compatibility of mission vehicles and depots, (4) develop initial depot concepts, (5) identify depot technology requirements, and (6) estimate non-recurring and recurring depot facility costs.

Orbital Propellant Depot Study Objectives

- Review Orbital Depot Requirements
- Perform Preliminary Trade Evaluations
  - Depot Location
  - SSF or Free Flyer Accommodations
  - Propellant Transfer Methods
- Mission Integration and Manifesting
- Propellant Depot Concepts
  - LEO Propellant Depot Concepts
  - Mars Propellant Depot Concepts
  - On-Orbit Safety Hazards Evaluation
- Depot Technology Assessment
- Depot Cost Assessment

SECTION 2

REQUIREMENTS ASSESSMENT

The following requirements taken from the Space Transfer Concepts and Analysis for Exploration Missions third quarterly review by Boeing Aerospace and Electronics, 22 June 1990 were assessed and updated where appropriate. Revisions, annotations and comments relative to the requirements are highlighted with bold letters, otherwise the requirements are deemed valid for this study. The reference in brackets refers to the view graph chart in the third quarterly review package. Vehicle configurations are shown in the charts "Cryo/AB Reference Configuration" and "Nuclear Thermal Rocket (NTR) Configuration".

(1) Mars Transfer System (MTS) [VG 2-01]

- (a) All passive cryogenic control system.
- (b) No MTV-TMIS fluid transfer before earth departure. (MEV tanks refrigerated or filled after MOI). [VG 2-02]

(2) Mars Transfer Vehicle-Trans-Mars Injection Stage (MTV-TMIS) [VG 2-02]

- (a) Passive thermal control system including zero-g thermodynamic vent system coupled to multiple vapor cooled shields.
- (b) TMIS insulating system is a continuously purged MLI over foam design optimized for minimum ground-hold, launch, and orbital boil-off. Includes vapor cooled shield (coupled to TVS) outside of foam.
- (c) TMIS tanks launched late in assembly sequence to minimize orbital stay time before TMI burn **(6 months - This orbital stay time increased because 90 day ETO launch centers were used where possible).**

(3) Mars Excursion Vehicle (MEV) [VG 2-03]

- (a) Passive cryogenic storage system: MLI with vapor cooled shields.
- (b) Gravity field environment eliminates need for zero-g acquisition and venting.
- (c) Vacuum jacketed ascent tanks for Mars boiloff reduction.

(4) Reference Cryo/Aerobrake Configuration [VG 3-01]

(a) A core stage with "plug-in" propellant tanks. Tanks and core stage rendezvous and dock automatically. Core stage provides simple plumbing. Vehicle assembled in SSF orbit.

(b) MTV prop. 85,141 kg.

(c) TMI prop. 490,950 kg, inert stage 54,560 kg. **Six** liquid hydrogen/LOX tank sets **(five plus the core)** each 7.4 m dia. x 15 m w/ shielding.

(5) Nuclear Thermal Rocket [VG 3-10]

(a) Vehicle assembled in SSF orbit. LH2 propellant tanks.

(b) *Earth Orbit Capture (EOC) prop. 27,756 kg,*
Trans Earth Inject (TEI) prop. 59,245 kg.
One EOC/TEI common tank 10 m dia. x 19m, 13,845 kg.

(c) *Mars Orbit Capture (MOC) prop. 151,680 kg.*
Two tanks 10 m dia.x 17 m, 25,572 kg.
(These tanks were reduced to a one tank configuration with a mass value equal to 60 mT).

(d) Trans Mars Inject (TMI) prop. *286,146 kg. Two tanks 10 m dia. x 30 m,* 43,092kg. **(Valid for missions one through four only).**

Mission Model

The following mission model has been assumed for the trade study evaluation.

(a) Seven missions with *five non-reusable tank sets per mission.* **(This was assessed to be three tank sets for the minimum science missions and six tank sets for the full science missions).**

SECTION 2

REQUIREMENTS ASSESSMENT

Cryo/AB Reference Configuration

The reference configuration shown is taken from the STCAEM third quarterly review. The configuration evaluated has been revised to include six liquid hydrogen/LOX tank sets (five plus the core).

Cryo/AB Reference Configuration

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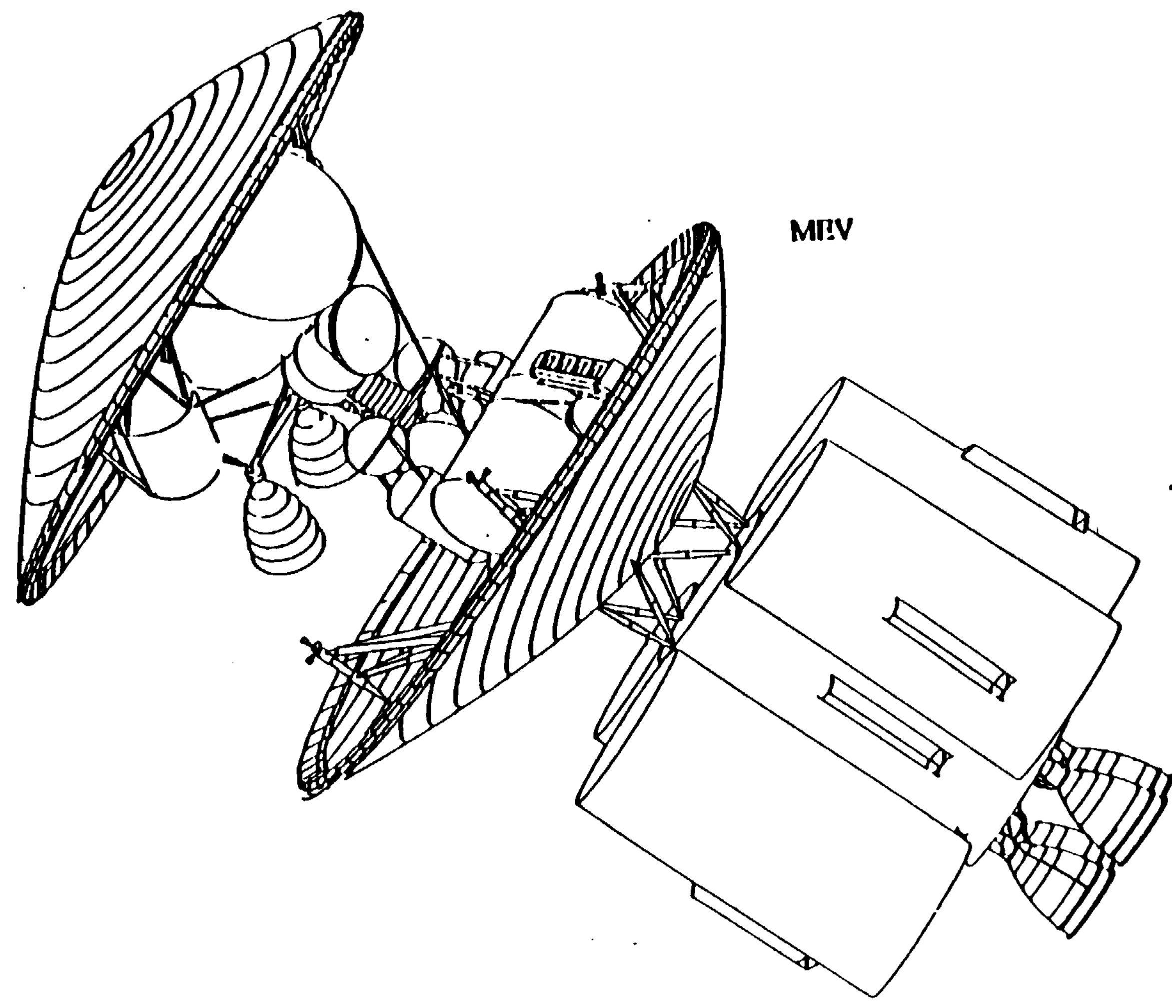

Nuclear Thermal Rocket (NTR) Configuration

The reference configuration shown is taken from the STCAEM third quarterly review. The configuration evaluated has been revised to the tank set configuration in the text.

Nuclear Thermal Rocket (NTR) Configuration

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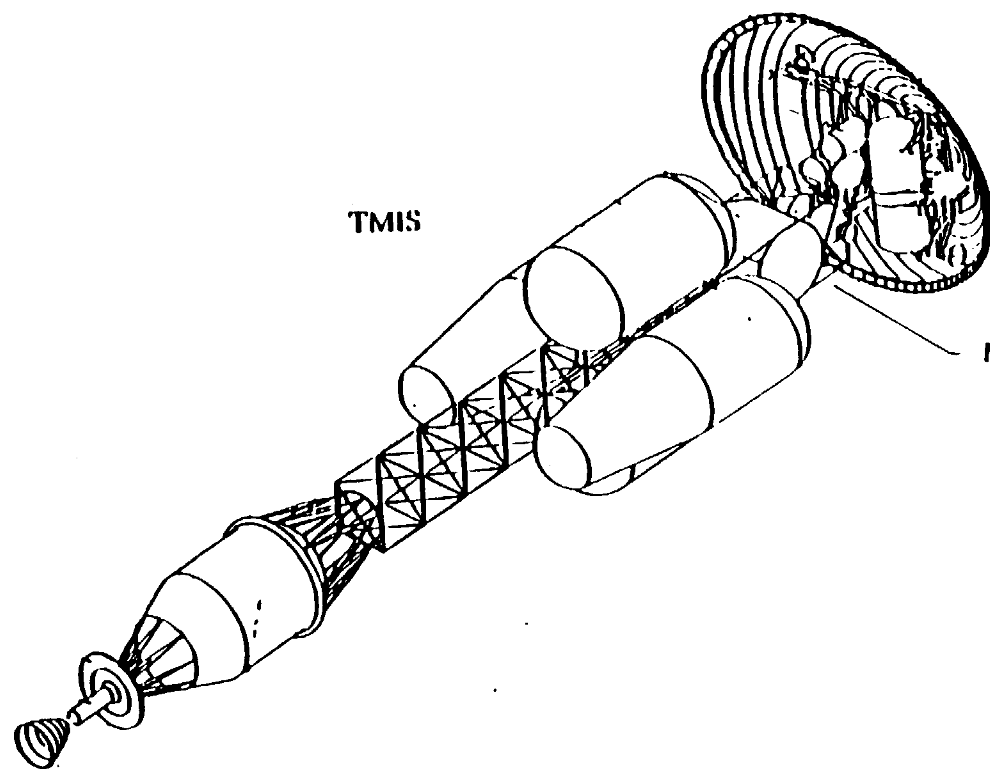


SECTION 3

ORBITAL PROPELLANT DEPOT TRADES

This section reports on the orbital propellant depot trade evaluations of operational location, accommodation selection (Space Station Freedom verses free flyer), and propellant transfer methods. Passive verses active refrigeration was previously evaluated under the Long Term Cryogenic Storage Facility Study and not repeated under this contract.

3.1 OPERATIONAL PROPELLANT DEPOT LOCATION

Of the five locations identified for the cryogenic depot, LEO represented the lowest mass to orbit and lowest operational complexity of the choices of LEO, GEO, Libration Point 2, and LLO. A summary of the potential depot locations is shown in the "Depot Location Sensitivity". The reduced vehicle mass resulting from higher launch orbits is nonetheless too small to balance out the large increase in ETO Mass required to ferry MTS and depot components, the result being an increase in overall ETO requirements. With no assumption of Lunar Propellant Production, Lunar vicinity locations lose their appeal due to the above reasoning. The complexity inherently involved in a split mission, LMO or Mars surface location for a cryogenic depot requires more detailed attention and has been covered in a separate section of this report.

Taking the above reasoning into account and allowing for man-tended personnel requirements of a cryogenic depot, a SSF co-orbiting altitude in LEO was chosen to be the best suited target location for a baseline of the depot trade. The added burden of establishing risk due to and necessary shielding required against micrometeoroid and orbital debris was recognized and has been detailed in a following section illustrating shielding requirements.

3.2 SPACE STATION FREEDOM OR FREE FLYER DEPOT ACCOMMODATIONS

3.2.1 Introduction

Given the co-orbiting nature of the baseline depot location, two possibilities exist: co-locating the node on SSF or utilizing a free-flying, co-orbiting depot. The issues associated with each are given in the chart "SSF Located vs. Separate Free Flyer". Given the stringent requirements for SSF microgravity experimentation and current efforts on the downsized space station configuration, the large masses reflected in the depot component of the vehicle propellant requirements point to the infeasibility of co-locating depot tanks on SSF. Vibrational disturbances of both high and low frequency occurring during mating of the depot tanks, as well as the comparable mass of the tanks with respect to station's downscaled overall mass, add severe operational complexity to SSF operations and may also lead to control problems due to the large C.G. shift caused by incorporation of the tanks.

A separate co-orbiting free flyer configuration would clearly add complexity to the infrastructure as the depot would incur separate power, reboost/deboost, and GN & C requirements. These components have been incorporated into the model manifests. Additional requirements are imposed by the co-orbiting nature of the concept. With respect to robotic operations, line-of-sight between SSF and the depot must be maintained. Man-tended operations require that the depot be located within the range of STS or other SSF-based personnel vehicles.

Due to the higher density and reduced cross-section of the depot configurations with respect to SSF, the ballistic behavior of a free flying depot poses the need for deboost as well as reboost capability. The faster decaying orbit of SSF would cause it to fall and accelerate away from a depot without the ability to pace. The option exists for using a greatly reduced aerobrake or other drag-inducing device which could be actively controlled to match orbital decay rates with SSF, thus saving propellant aboard the depot and possibly eliminating the need for reboost capability of the depot, relying instead on LTV or similar vehicles to provide the necessary reboost thrust during SSF reboost phase.

3.2.2 Orbital Debris Environment

With the selection of LEO for the SSF, co-orbiting depot location is linked the burden of identifying the orbital debris and micrometeoroid environment and determining the resultant risk to space structures so that they may be sufficiently shielded from such threats. The orbital debris environment is summarized in the charts "Orbital Debris Environment".

Although traveling at hypervelocities (~20 km/s), micrometeoroid particles do not pose the largest threat to the integrity of space structures. Their relatively small size and mass overshadow the high speeds with which they impact objects in orbit, contributing mostly to pitting or abrasion of protective coatings. Their random entry into earth atmosphere makes for an evenly distributed fluence, with no one orientation seeing a concentration of impacts.

Orbital debris on the other hand poses a greater risk. Originating from intentional or accidental fragmentation of payloads, spent rocket bodies, or jettisoned payload equipment, these large particles carry sufficient velocity (~7 km/s) and mass to cause serious damage if not properly protected against. In a report on orbital debris by the National Security Council, data from ground based telescopes, shuttle data, and recovered satellites (Solar Max, Palapa B) were used to model the micrometeoroid and orbital debris environment in LEO. Based on current launch rates, a model for the orbital debris environment in 2010 was extrapolated, revealing the predominance of the orbital debris threat over that of the micrometeoroid threat.

A range of debris sizes from 1-4 cm was identified as the most threatening. Above or below these sizes, methods to counter damage are through collision avoidance maneuvers or shielding, respectively. However, due to the uncertainty inherent in current debris cross-sectional area measurements, an order of magnitude uncertainty in particle size was considered appropriate to determine impact probability. At the lower end of the range, the frequency of 0.1 cm objects striking a space structure located in LEO was determined to be .02 impacts per square meter • year, which is the rate employed in determining impact risk to vehicles during assembly.

3.2.3 Impact Risk

Given the configuration of the Cryogenic All Propulsive (CAP) and Nuclear Thermal Rocket (NTR) Conjunction Class Mars Transfer Vehicles which a depot might support, overall cross-sectional areas presented by the vehicle tanks were calculated and the probability of impact was determined based on the highest fluence found for particles in the .1-4 cm range, this being .02 impacts per square meter • year. This is shown in the chart "Impact Risk".

Based on the assumption of 90 day launch centers for delivery missions of vehicle components, the long duration on orbit leads to a good probability of impacts over the assembly life of the vehicle. Implications of such an impact on an unprotected structure are serious. Penetration resulting in loss of propellant could spawn loss of vehicle control due to uncontrolled thrusting or loss of crew member(s) due to suit contamination. These issues are covered explicitly in the section on safety issues.

Impact risk drives the depot configuration as shown in the chart "Shielding Considerations".

3.3 PROPELLANT TRANSFER METHODS

3.3.1 Introduction

The advantages and disadvantages of transferring cryogenic propellants in zero-g and artificial-g environments are explored in this trade study. The "Zero-g Versus Artificial-g Liquid Transfer Trade Tree" outlines this trade study. Liquid hydrogen and liquid oxygen are the propellants considered (Cryo/Aerobrake configuration). The sensitivity of the results to only liquid hydrogen transfers (Nuclear Thermal Rocket configuration) is also examined. There are five main areas involved in liquid transfers: liquid acquisition, receiver tank childown, receiver tank filling, transfer method, and pressure control. The "Liquid Transfer Options Versus Depot Concept" shows that the technique selected for each area is specific to the depot concept. There are a number of techniques for acquiring liquid in zero-g, so these are traded first to obtain the "best" zero-g system. This "best" system is then traded against other types of liquid transfer which are germane to the particular depot concept.

3.3.2 Propellant Acquisition Trade Analysis

3.3.2.1 Requirements. The primary requirement for this trade is that pure liquid be supplied at the tank outlet while in a zero-g environment. The fluids to be transferred are liquid oxygen and liquid hydrogen. The supply tanks are assumed to be large cylindrical type (>3m diameter).

3.3.2.2 Description of Candidate Design Approaches. The zero-g propellant acquisition candidates can be grouped into three broad classes: surface tension, positive expulsion, and other. Surface tension devices include screened channels, single or double screen tank linings, and perforated plates. These are commonly referred to as Liquid Acquisition Devices (LADs). These devices rely on the surface tension to wick only liquid to a tank outlet. Positive expulsion devices include bladders, bellows, pistons, and diaphragms. The differences between bladders, bellows, and diaphragms can be seen in "Three Types of Positive Expulsion Systems". All positive expulsion devices physically move a barrier to expel liquid. Other devices includes a wide range of systems, such as fluid rotation (by paddle or tangential jets), tank rotation, dielectrophoresis (which relies on the dielectric properties of the fluid to orient liquid and vapor within an electric field), and acoustic/magnetic devices.

3.3.2.3 Comparison of Alternative Approaches. Initial screening of the candidate approaches eliminated all but the five shown in "Propellant Acquisition Trade". Piston devices were eliminated because of their inherent high weight and problems associated with their moving cryogenic seals. Dielectrophoresis was eliminated because no operational systems are available (although successful tests were completed with Freon 113 and LN2 in KC-135 flights). Also further work needs to be done to ensure safety for use with LO2. Acoustic/magnetic devices were deleted from further consideration because attempts at demonstrating feasibility were unsuccessful.

"Propellant Acquisition Trade" shows the results of the propellant acquisition trade. Ratings of 1–5 were used with 1 being the best. Surface tension devices (LADs) are a clear winner, but the other candidates are closely scored. Transfer time didn't turn out to be a discriminator because they all can expel liquid at reasonable rates.

The primary advantages of surface tension devices over the other systems are their low weight, simplicity, fluid compatibility, and long useable lifetime. Although their use has not yet been demonstrated with cryogenic fluids in zero-g. This presents some development risk since these devices fail when a portion of the screen unwets and vapor can be drawn into the tank outlet. Screen unwetting with cryogens can be caused by heat transfer to screens which vaporizes liquid.

Bladders provide a physical barrier between the liquid and the pressurant. They are a relatively simple system. Data from previous development work (Reference 1) indicates that collapsing bladders are preferable to the expanding type. In the collapsing bladder system, the bladder collapses around a perforated standpipe. Bladders have been used successfully in non-cryogenic applications. Material compatibility problems present a large development risk for use with LO2 and LH2. Materials which remain flexible at cryogenic temperatures are not completely safe with LO2. A problem with using these bladders with LH2 is that the hydrogen can permeate the plies and cause delamination when warmed back to ambient temperature. Some recent work has been done with aluminum bladders for non cryogenic fluids (Reference 6). If these could be applied to cryogens, they could solve most of the problems.

Bellows can be cycled a large number of times (~1000 cycles) without fatigue especially at cryogenic temperatures. Another advantage of this system is that it eliminates the need for a zero-g mass gage because the amount of liquid can be correlated with the position of the bellows. A major disadvantage of bellows is that they are heavier than other candidates. Manufacture of large (> 1m) diameter bellows presents a significant development risk.

The diaphragm considered in this trade is a metallic reversing hemisphere. The primary advantage of this system is the low residuals (the lowest of the five systems considered in this trade). The main disadvantage is the low number of reuse cycles (5–10 cycles).

The fluid rotation system considered for this trade uses a rotating paddle rather than tangential jets. The main advantage this system is the positive positioning of the liquid so that mass gaging and venting systems can be easily incorporated. The primary disadvantages are the need for a motor drive system and the high residuals. Also this system is better suited to spherical tanks.

3.3.2.4 Sensitivities. If LH2 was the only propellant as in the Nuclear Thermal Rocket configuration, the material compatibility problems associated with LO2 would be eliminated. However, the surface tension system would still be the preferred option.

3.3.2.5 Conclusion and Recommendations. The surface tension device system is recommended as the best candidate for zero-g liquid acquisition. Zero-g, cryogenic testing/demonstration of LADs are required prior to use in the Advanced Space Transportation Vehicle. The Cryogenic On-Orbit Nitrogen Experiment (CONE), an STS flight experiment scheduled for 1995, will be testing LADs with LN2. This will hopefully provide enough data to verify models and give confidence to LH2 and LO2.

3.3.3 Depot Concept Trade Analysis

3.3.3.1 Requirements. The requirements for this trade are the same as those for the propellant acquisition trade. Namely that pure liquid be supplied at the tank outlet while in a zero-g environment. The fluids to be transferred are liquid oxygen and liquid hydrogen. However, there are not any constraints on the size or geometry of the supply tanks.

3.3.3.2 Description of Candidate Design Approaches. The four depot concepts considered in this trade are: non propulsive (zero-g), linear propulsive, rotating propulsive and tether. As shown in "Liquid Transfer Options Versus Depot Concept", the latter three concepts all use the same techniques for receiver chilldown, receiver filling, and pressure control. The only difference being that the rotating propulsive concept would not require pumps for transfer.

The non propulsive concept is our baseline and is shown in "Depot Concept for Support of CAP Vehicle" and "Depot Concept for Support of NTR Vehicle" This baseline concept is essentially a truss structure with large cylindrical tanks. The tanks contain liquid acquisition devices (LAD's). This passive depot concept was studied extensively under the Long Term Cryogenic Storage Facility Systems Study.

The linear propulsive concept consists of a structure with tanks mounted on it. Thrusters are mounted on the structure to provide a linear thrust. The liquid can then be settled to one end of the tanks. Settling would be required to acquire liquid from the supply tank and prior to any venting.

The rotating propulsive option consists of toroidal tanks which rotate about their centers. Thrusters are required to provide rotation which forces the liquid to the outside of the tori. By positioning the receiver tank at a radius greater than the radius of revolution of the torus, transfer can occur without a pump.

The tether concept relies on the gravity gradient along a radius from the earth. If the depot has tanks that are separated by a sizeable distance (e.g. dumbbell), then the axis of the depot will align with the Earth's radius and will orbit at a velocity of the depot's center of mass. Since the liquid in the tank closest to the earth is traveling at a velocity less than that required to keep it in orbit at this distance from the Earth, the liquid will be pulled toward the Earth. Similarly, in the tank furthest from the Earth, the liquid is travelling faster than required to keep it at that distance from the Earth so it is pulled away from the Earth. Thus, the liquid settles away from the center of mass of the depot and settled operations can be performed.

3.3.3.3 Comparison of Alternative Approaches. The amount of propellant used with the propulsive options is dependent on the number of transfers and ventings that are required. These are in turn dependent on the number of missions, tanker capacities, etc. However, the propulsive option presents the least technical risk as far as the fluid processes are concerned. The rotating depot has the advantage that pumps are not required but there are a lot of other technical risks associated with this option.

The tether concept requires large tether lengths, for example, an artificial gravity of 10-3 g's would require a tether length of 1.4 nautical miles from the center of gravity. The main advantages of this concept are that it is passive and that settled chilldown, transfer, and pressure control techniques can be used.

3.3.3.4 Recommendation. Further study is required to determine which of the concepts is best overall. However, the baseline concept presented elsewhere is the non propulsive (zero-g) system due to the extensive study that this concept received under the Long Term Cryogenic Storage Facility Systems Study. This trade analysis is summarized in the chart "Depot Concept Trade Analysis".

3.3.4 References

The references given below are bibliographical and are not all called out in the text.

[1] Stark, J.A., "Study of Low Gravity Propellant Transfer – Final Report," GDCA-DDB-72-002, NAS8-26236, June 1972.

[2] Stark, J.A., "Low-g Fluid Transfer Technology Study – Final Report," CASD-NAS-76-014, NAS3-17814, May 1976.

[3] Kroll, K., "Propellant Transfer – Tethered Depot," NASA JSC, N8517006.

[4] "Long Term Cryogenic Storage Facility Systems Study – Interim Report," Vol. I Part B – Study Results, GDSSD-SP-86-038, NAS8-36612, October 1986.

[5] Bennett, F.O., et al., "Space-Based Propellant Management Systems," GDC-ERR-84-456, December 1984.

[6] Biron, J., "An Aluminum Collapsible Bladder Tank for Space Systems," AIAA 90-2058, July 1990

SECTION 3

ORBITAL PROPELLANT DEPOT TRADES

Depot Location Sensitivity

Large radiation and debris hazards occupy altitudes of 500-20000 km. Realistic orbits are LEO, GEO, L2 and LLO.

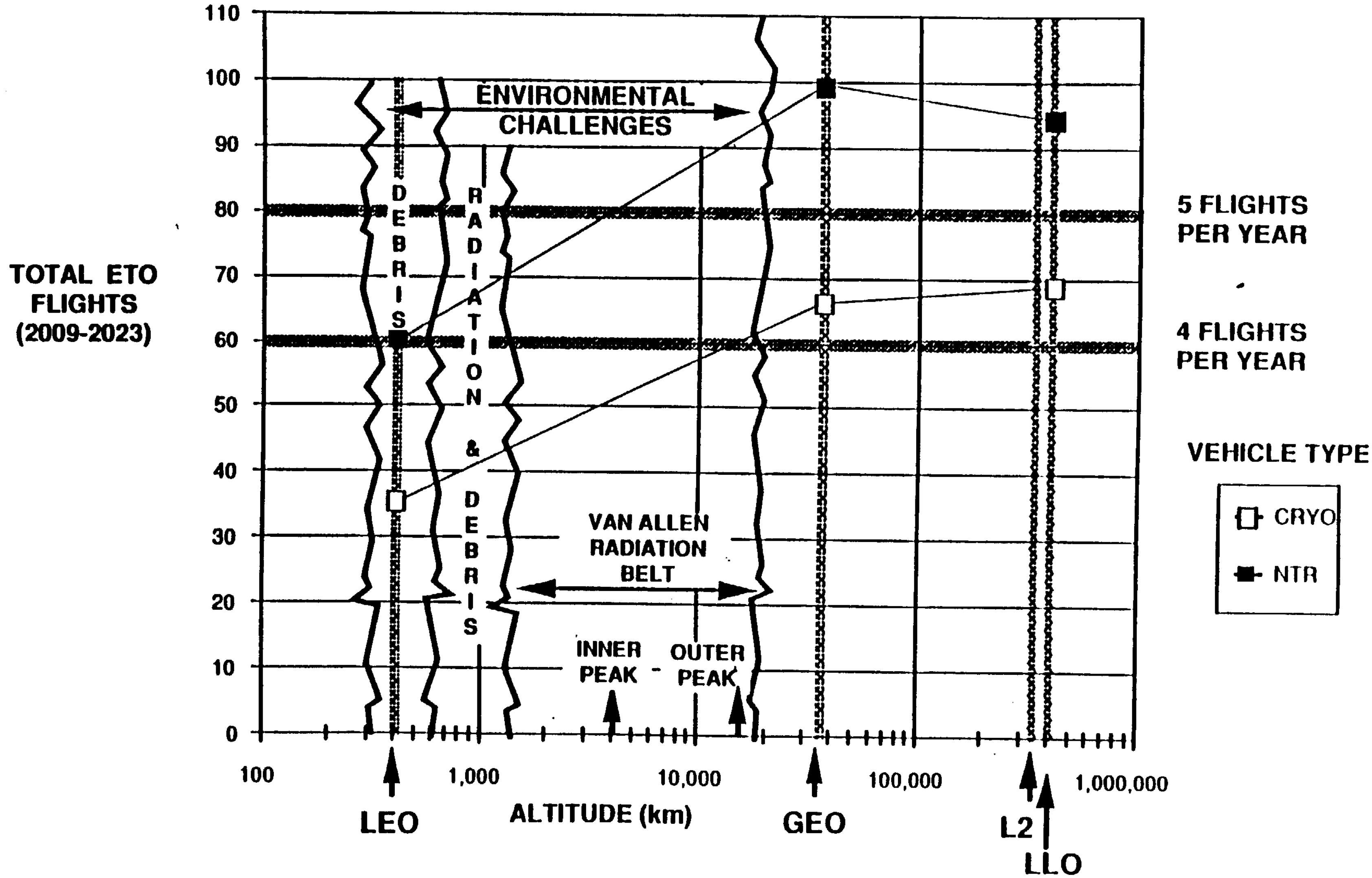
With no Lunar Propellant Production, no savings in going to Lunar orbit.

GEO shows highest performance demand ≈ highest cost. Reduced vehicle mass resulting from higher launch orbit not sufficient to balance out large increase in ETO mass required to ferry MTS and depot components out of Earth's gravity well, resulting in increase in overall mass to ETO.

Only LEO satisfies four flights per year (90 launch centers) criteria.

Orbital debris still remains a hazard in LEO, which must be understood and contended with.

Depot Location Sensitivity



SSF Located vs. Separate Free Flyer

<u>SSF Issues</u>

Disruption of the micro-g environment as well as SSF control problems due to large C.G. shifts are likely as a result of incorporating large tanks into the SSF architecture

Suggesition: Could be used as an initial deployment for depot and vehicle trusses.

<u>Free Flyer Issues</u>

A remote depot location incurs separate power, reboost / deboost, and GN & C capabilities

Line-of-sight must be maintained for robotic communications in assembly and monitoring during man-tended periods.

Depot must remain within range of SSF for crew transfer.

Suggestions: Aerobrake could reduce propellant requirements sufficiently to irradicate need of reboost thrusters

SSF Located vs. Separate Free Flyer

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SSF Issues

- Vibrational disturbances of micro-g environment
- Control problems likely due to large C.G. shift

Suggestions:

- Possible initial deployment of depot truss structure based at SSF

Free Flyer Issues

- Configuration incurs separate power, reboost / deboost, and GN & C capability
- Line-of-sight must be maintained concerning robotics
- Must remain within range of STS or other personnel vehicle from SSF

Suggestions:

- Downscaled aerobrake or drag inducing device actively controlled to facilitate minimal propellant paced descent along with SSF

Orbital Debris Environment

Orbital debris environment at 500 km x 28.5° orbit is dominant threat--micrometeoroid threat relatively small

Model comes from data gathered by National Security Council from several sources:

- -Ground based telescopes
- -STS data
- -Recovered satellites (Solar Max, Palapa B)

Orbital Debris Environment

Model based on environment existing in 500 km x 28.5° orbit

Impact events dominated by orbital debris; micrometeoroid threat smaller by an order of magnitude

Concerned with range of debris between 1cm and 4cm cross-section

Objects larger than 4cm in cross-section trackable; can utilize collision avoidance

Uncertainty in debris cross-sectional area measurements on order of one magnitude; must take into account range of debris between 0.1-4 cm

Highest threat seen to be .02 impacts per $m^2 \cdot yr$

Orbital Debris Environment (cont'd)

Cross-sectional area measurement uncertain by order of magnitude.

Must increase range of concern to 0.1--4.0 cm.

Highest threat seen to be .02 impacts / square meter • year.

Orbital Debris Environment (cont'd.)

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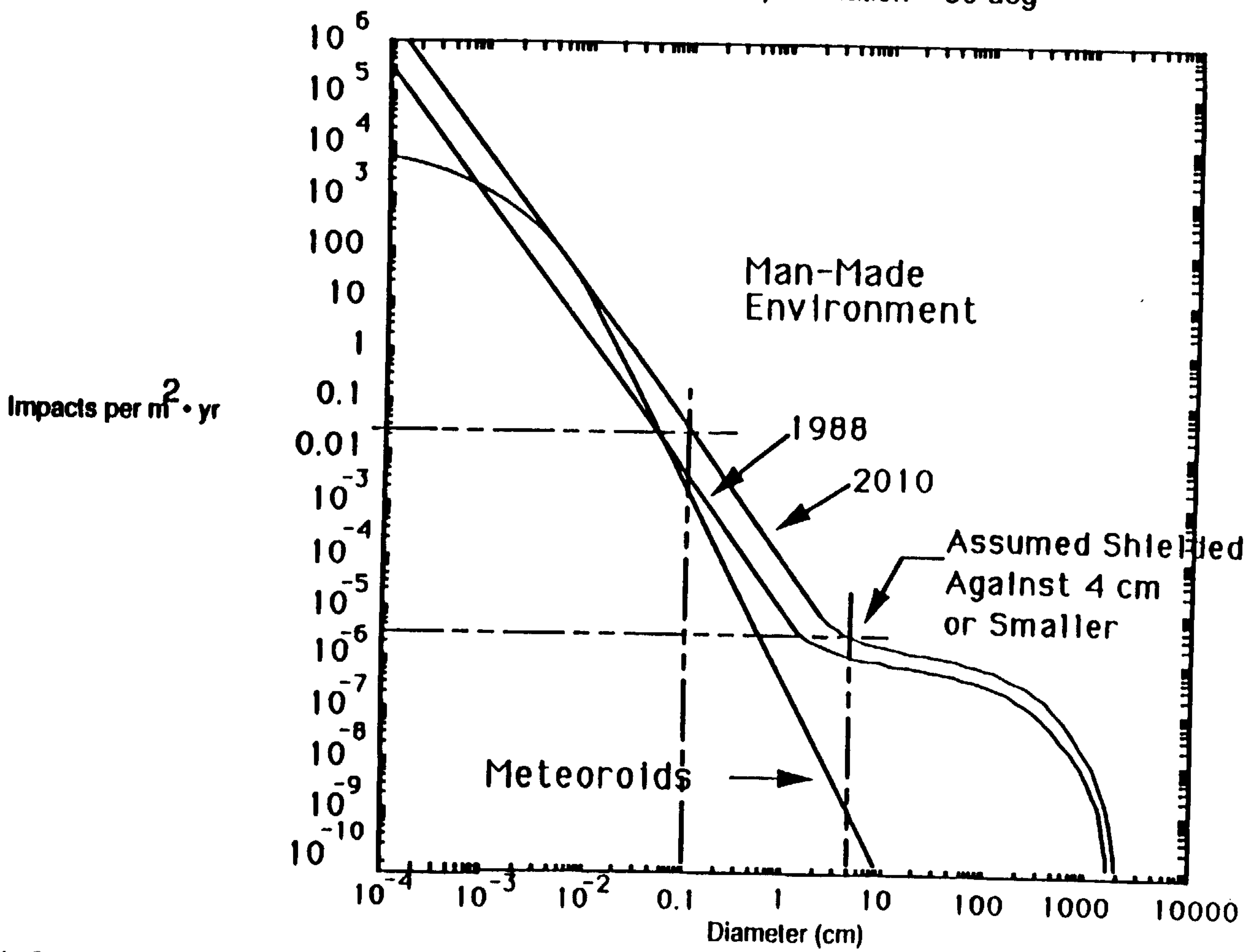
Impact Rates on Large Space Structures *

altitude = 500 km; inclination = 30 deg



* Source: Report on Orbital Debris, National Security Council, 1988.

VG3-04

Impact Risk

Determined exposed cross-sectional area of vehicle tanks for CAP and NTR configurations.

Based on impact frequency of orbital debris, determined probability of impacts per year

Long duration on orbit of tanks due to 90 launch centers leads to good probability of impacts without proper shielding

Implications:
- -Loss of vehicle due to control loss
- -Loss of crew member(s) due to suit contamination

Impact Risk

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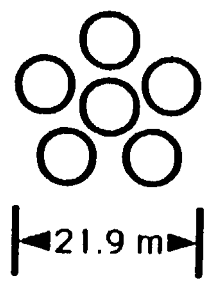
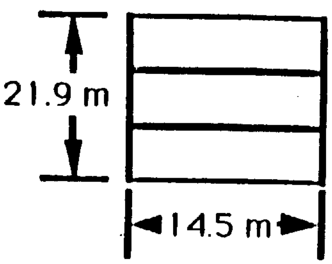
All Propulsive

7-8 strikes/year



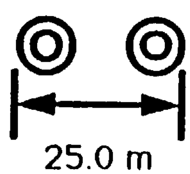
$SA = 375.8\ m^2$



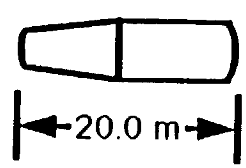
$SA = 317.6\ m^2$

NTR Vehicle

4-5 strikes/year



Surf. Area = $250.0\ m^2$



Surf. Area = $200.0\ m^2$

- Based on rate of .02 strikes/m^2 • yr
- Assumed worst case flux of debris: 1 mm to 4 cm
  - < 0.1 cm--non-critical impact
  - > 4 cm--collision avoidance initiated
- Assumed current launch rate debris growth predictions for 2010
- Baseline of 90 day launch centers leads to good probability of impacts over assembly life of vehicle
- Implications:
  - -Loss of vehicle control
  - -Loss of crew member(s) due to suit contamination

VG3-05

Shielding Considerations

Preliminary LDEF results confirm NASA debris model:

- -20 times more impacts on leading edge than trailing edge.
- -10-20 times more impacts on space end.
- -Nearly all impacts reflect 2 deg incidence ban with local horizon.
- -Bulk of impacts due to orbital debris impingement.

Fluence of orbital debris largely contained within 45 ° body angle in plane of local horizon.

To minimize shield surface area, weight, tanks should be nested in cubic formation.

To minimize tank weight penalty, ease deployment, and remove tank wall from debris shield system, the shielding has been kept separate from the tankset.

Shielding Considerations

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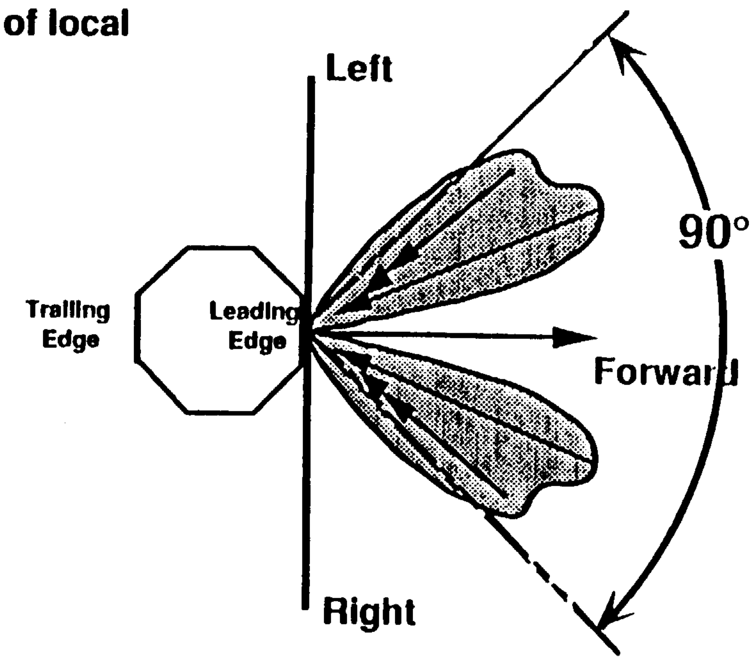
- Preliminary LDEF results back-up NASA debris model
  - 20 times more impacts on leading than trailing edge
  - 10-20 times more impacts on space end than on earth end
  - Trailing edge impacts result of micrometeoroid impacts
  - Nearly all debris impacts occurred within 2 deg of local horizon
- Fluence field largely contained within 45° solid angle
- Drives tanks towards cubic configuration to minimize shielding surface area and weight
- Shielding is kept separate from tankset to minimize weight penalty, ease deployment



Zero-g Versus Artificial-g Liquid Transfer Trade Tree

The advantages and disadvantages of transferring cryogenic propellants in zero-g and artificial-g environments are explored in this trade study. Liquid hydrogen and liquid oxygen are the propellants considered (Cryo/Aerobrake configuration). The sensitivity of the results to only liquid hydrogen transfers (Nuclear Thermal Rocket configuration) is also examined.

The trade tree shows the different options that are considered in this trade study. The "best" zero-g acquisition device will be traded against the artificial-g concepts. The depot concepts considered are described below.

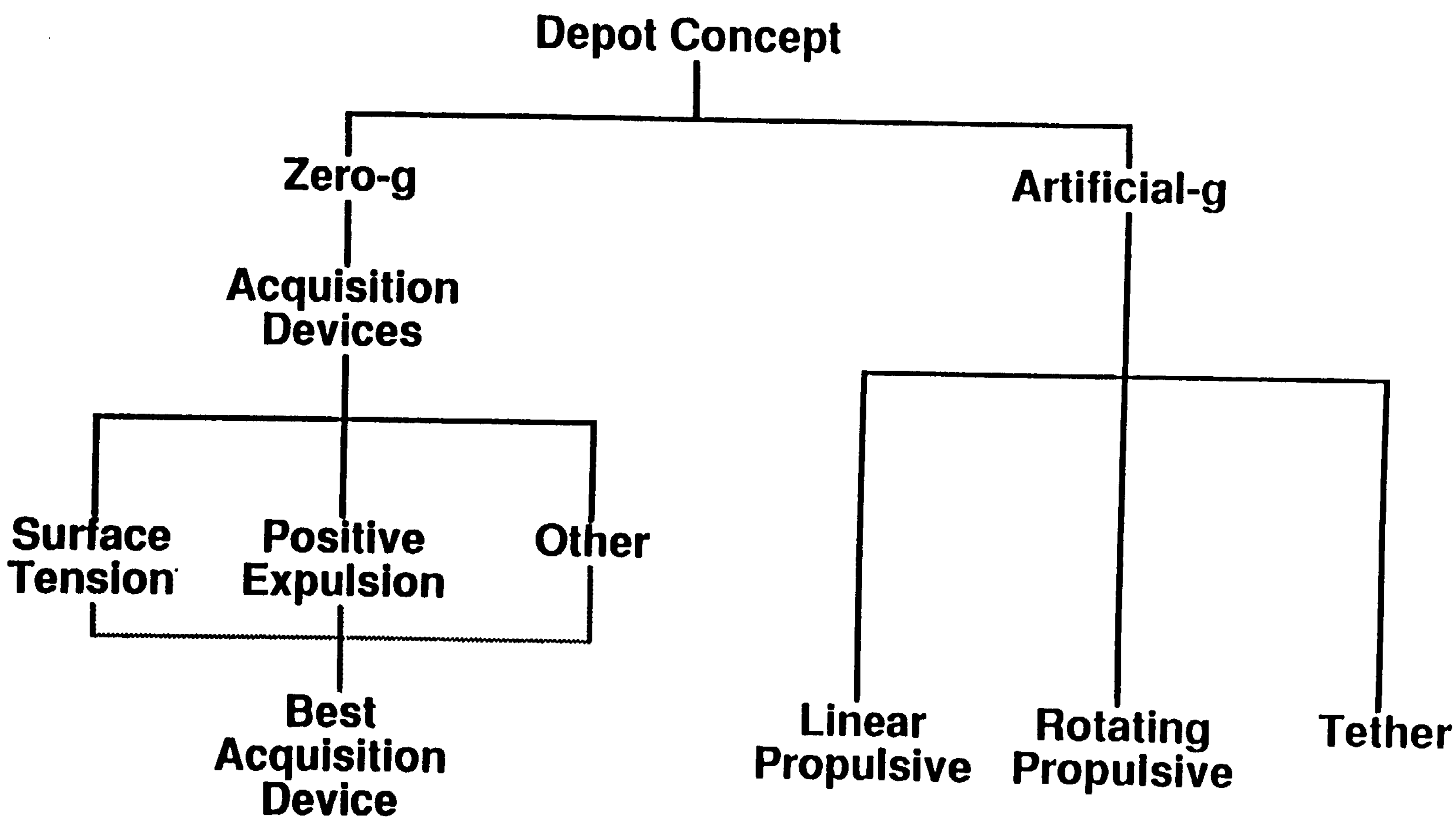
The non propulsive concept is essentially a truss structure with large cylindrical tanks. The tanks contain zero-g liquid acquisition systems.

The linear propulsive concept consists of a structure with tanks mounted on it. Thrusters, mounted on the structure, provide a linear thrust. Liquid can then be settled to one end of the tanks. Settling would be required to acquire liquid from the supply tank and prior to any venting.

The rotating propulsive option consists of toroidal tanks which rotate about their centers. Thrusters provide rotation which forces the liquid to the outside of the tori. By positioning the receiver tank at a radius greater than the radius of revolution of the torus, transfer can occur without a pump.

The tether concept uses the gravity gradient to orient the tanks along a radius from the earth. The depot then orbits at the velocity of it's center of mass. Liquid settles away from the center of mass of the depot and settled operations can be performed.

Zero-g Versus Artificial-g Liquid Transfer Trade Tree



Liquid Transfer Options Versus Depot Concepts

There are five main areas involved in liquid transfers: liquid acquisition, receiver tank chilldown, receiver tank filling, transfer method, and pressure control. The technique selected for each area is specific to the depot concept. There are a number of techniques for acquiring liquid in zero-g, so these are traded first to obtain the "best" zero-g system. This "best" system is then traded against other types of liquid transfer which are germane to the particular depot concept.

Of the four depot concepts, the linear propulsive, rotating propulsive and tether concepts all use the same techniques for receiver chilldown, receiver filling, and pressure control. The only difference being that the rotating propulsive concept would not require pumps.

Liquid Transfer Options Versus Depot Concepts

	DEPOT CONCEPT			
	Non Propulsive	Propulsive (Linear)	Propulsive (Rotating)	Tether
Liquid Acquisition				
Surface Tension	√			
Positive Expulsion	√			
Other Zero-g	√			
No Special Requirement		√	√	√
Receiver Chilldown				
Charge/Hold/Vent	√			
Wall Heat Exchanger	√			
Settled Chill (Vent to Space)		√	√	√
Receiver Filling				
No-Vent Fill	√			
Ullage Exchange	√			
Settled Fill (Vent to Space)		√	√	√
Transfer Method				
Helium Pressurization	√	√	√	√
Autogenous Pressurization	√	√	√	√
Pump w/ Helium Pressurization	√	√		√
Pump w/ Autogenous Press'n	√	√		√
No Special Requirement			√	
Pressure Control				
Thermodynamic Vent System	√			
Settled Vent		√	√	√

Three Types of Positive Expulsion Systems (Propellant Acquisition Trade Analysis)

Requirements

The primary requirement for this trade is that pure liquid be supplied at the tank outlet while in a zero-g environment. The fluids to be transferred are liquid oxygen and liquid hydrogen. The supply tanks are assumed to be large cylindrical type (>3m diameter).

Description of Candidate Design Approaches

The zero-g propellant acquisition candidates can be grouped into three broad classes: surface tension, positive expulsion , and other. Surface tension devices include screened channels, single or double screen tank linings, and perforated plates. These are commonly referred to as Liquid Acquisition Devices (LADs). These devices rely on the surface tension to wick only liquid to a tank outlet. Positive expulsion devices include bladders, bellows, pistons, and diaphragms. The differences between bladders, bellows, and diaphragms are shown here. All positive expulsion devices physically move a barrier to expel liquid. Other devices includes a wide range of systems, such as fluid rotation (by paddle or tangential jets), tank rotation, dielectrophoresis (which relies on the dielectric properties of the fluid to orient liquid and vapor within an electric field), and acoustic/magnetic devices.

Comparison of Alternative Approaches

Initial screening of the candidate approaches eliminated all but the five shown here. Piston devices were eliminated because of their inherent high weight and problems associated with their moving cryogenic seals. Dielectrophoresis was eliminated because no operational systems are available (although successful tests were completed with Freon 113 and LN2 in KC-135 flights). Also further work needs to be done to ensure safety for use with LO2. Acoustic/magnetic devices were deleted from further consideration because attempts at demonstrating feasibility were unsuccessful.

Three Types of Positive Expulsion Systems

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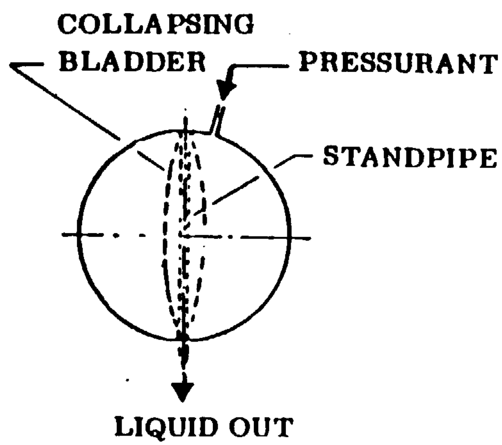
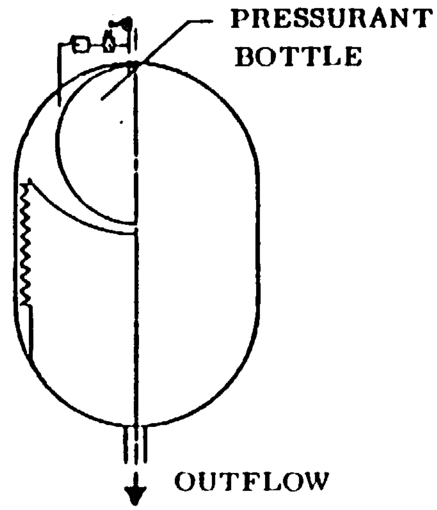
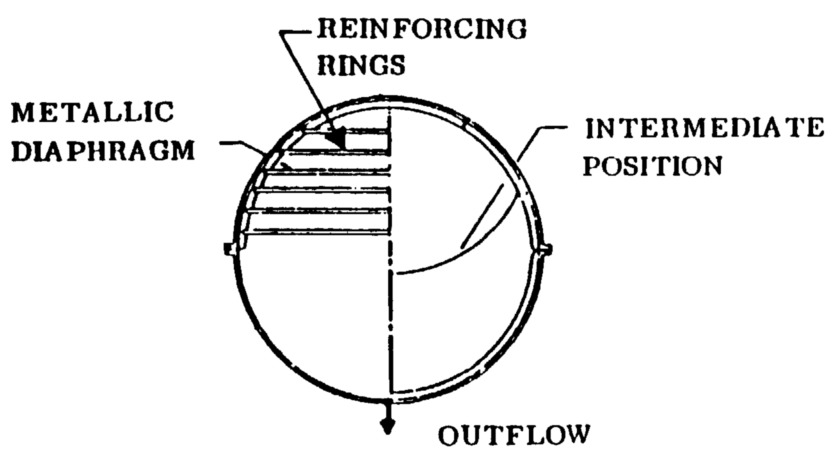


a. Bladder System



b. Bellows System



c. Diaphragm System

Propellant Acquisition Trade

Surface tension devices (LADs) are a clear winner, but the other candidates are closely scored.

Surface tension devices have low weight, simplicity, good fluid compatibility, and long useable lifetime. Their use has not yet been demonstrated with cryogenic fluids in zero-g. This presents some development risk since these devices fail when a portion of the screen unwets and vapor can be drawn into the tank outlet. Screen unwetting with cryogens can be caused by heat transfer to screens which vaporizes liquid.
Bladders are a relatively simple system, providing a physical barrier between the liquid and the pressurant. Collapsing bladders, where the bladder collapses around a perforated standpipe, are preferable to the expanding type . Bladders have been used successfully in non-cryogenic applications. Material compatibility problems present a large development risk for use with LO2 and LH2 since materials which remain flexible at cryogenic temperatures are not completely safe with LO2 and hydrogen can permeate the plies and cause delamination when warmed back to ambient temperature.
Bellows can be cycled a large number of times (~1000 cycles) without fatigue especially at cryogenic temperatures. Bellows eliminate the need for a zero-g mass gage but they are heavier than other candidates. Manufacture of large (> 1m) diameter bellows presents a significant development risk.
The diaphragm considered here is a metallic reversing hemisphere. This system has the the lowest residuals of the five systems considered but has a low number of reuse cycles (5–10 cycles).
The fluid rotation system considered uses a rotating paddle and requires a motor drive system. This provides positive positioning of the liquid so that mass gaging and venting systems can be easily incorporated but leaves high residuals. This system is better suited to spherical tanks.

If only LH2 was used, as in the NTR configuration, the material compatibility problems associated with LO2 would be eliminated. However, the surface tension system would still be the preferred option.

Recommendation
The surface tension device system is recommended as the best candidate for zero-g liquid acquisition. Zero-g, cryogenic testing/demonstration of LADs are required prior to use in the ASTV.

Propellant Acquisition Trade

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	Surface Tension	Bladder	Bellows	Diaphragm	Paddle
Development Risk	3	5	5	4	2
Weight	2	2	5	4	3
System Complexity	1	2	2	2	4
Useable Lifetime	1	3	2	4	1
Boil-off Penalty	1	1	1	1	3
Residuals	5	4	2	1	5
Unit Cost	2	2	5	4	2
Development Cost	3	4	5	4	3
Operational Complexity	2	2	2	2	2
Reliability	2	3	2	2	2
Power Requirements	1	1	1	1	3
Transfer Time	1	1	1	1	1
Total	24	30	33	30	31

Ratings of 1–5 with 1 being the Best

Depot Concept Trade Analysis

<u>Requirements</u>
Single phase liquid oxygen and liquid hydrogen must be supplied at the tank outlet while in a zero-g environment. There are no constraints on the size or geometry of the supply tanks.

<u>Description of Candidate Design Approaches</u>
The non propulsive concept (our baseline) is essentially a truss structure with large cylindrical tanks. Based on the results of the zero-g acquisition trade, a surface tension liquid acquisition system (liquid acquisition device, LAD) was chosen.

The linear propulsive, rotating propulsive and tether concepts are the same as were described earlier.

<u>Comparison of Alternative Approaches</u>
The propulsive options present the least technical risk as far as the fluid processes are concerned but, the amount of propellant used is dependent on the number of transfers and ventings that are required. These are in turn dependent on the number of missions, tanker capacities, etc. The rotating depot does not require pumps but has a lot of other technical risks. The tether concept requires large tether lengths but it is passive and settled chilldown, transfer, and pressure control techniques can be used.

<u>Recommendation</u>
Further study is required to determine the best overall system. The baseline non propulsive concept was chosen due to the extensive work performed under the Long Term Cryogenic Storage Facility Systems Study.

Depot Concept Trade Analysis

- The non propulsive concept (our baseline) uses a Liquid Acquisition Devices (LADs) inside of large cylindrical tanks.
- The linear propulsive, rotating propulsive and tether concepts are the same as were described earlier.
- The propulsive options present the least technical risk as far as the fluid processes are concerned but, the amount of propellant used is dependent on the number of transfers and ventings that are required. These are in turn dependent on the number of missions, tanker capacities, etc. The rotating depot does not require pumps but has a lot of other technical risks. The tether concept requires large tether lengths but it is passive and settled chilldown, transfer, and pressure control techniques can be used.
- Further study is required to determine the best overall system. The baseline non propulsive concept was chosen due to the extensive work performed under the Long Term Cryogenic Storage Facility Systems Study.

SECTION 4

MISSION INTEGRATION AND MANIFESTING

This section presents a summary of the exploration scenarios for the Mars mission models evaluated. Three fueling options were considered for each of the two Mars exploration scenarios. This analysis helps establish depot capacities and identifies top level trends when a depot is included in the LEO infrastructure.

4.1 EXPLORATION SCENARIOS

Two Mars mission models were supplied to use as references, they are termed the Minimum Science Scenario and the Full Science Scenario. These two models of extraterrestrial exploration differ greatly in scope, as their names might suggest. A comparison of the two scenarios will show exactly how the two vary in terms of human presence, technology required, strategy, and cost. The main points of this discussion are summarized in "Summary of Exploration Scenarios".

The Minimum Science Scenario is the less ambitious of the two, fulfilling a philosophy to visit diverse sites on the Martian surface for brief human exploration. It comprises three missions over the nine years from 2015 to 2023. The missions in each case are identical in operations and hardware, but visit different Martian sites. Each is a conjunction class mission carrying six crew. The Martian surface payload is delivered by two landers, and consists of crew accommodations for 30 days, two unpressurized rovers that can be operated manually or telerobotically, and other exploration tools. The MTV utilizes cryogenic liquid hydrogen (LH2) and liquid oxygen (LOX) propulsion, and is totally expended through staging during each mission. The crew eventually returns to Earth by Apollo-style reentry. This program is somewhat stand alone, meaning a non-reliance on any dedicated space infrastructure.

The Full Science Scenario, on the other hand, places an emphasis on the establishment of long term bases and extensive surface exploration. Two manned bases are founded during six missions that span from 2009 to 2023. Each mission is unique, and fulfills a particular step in the establishment, expansion, and consolidation of the bases. This model also used conjunction class missions, but the crew size increases from 6 to 12 on the fourth mission. In order to support all of this activity at Mars, the MTV utilizes more advanced Nuclear Thermal Rocket (NTR) propulsion, which is assumed to be mature and man-rated by 2009. The propellant for this propulsion technology is LH2 only. Another difference from the Minimum Science Scenario is that much of the MTV is refurbished and reused after its return to Earth vicinity. The reusable parts include the crew habitat, truss, reactor, engines, and shielding. For those missions which employ recycled MTVs, only the payload, MEVs, and crew need to be delivered to orbit. Upon capture at Earth, the NTR is inserted into a high orbit. An LTV must act as a ferry, bringing propellant to the NTR for the return to LEO, and retrieving the crew returning from Mars.

4.2 FUELING OPTION EVALUATIONS

A major consideration in the assembly and preparation of an MTV is to deliver the propellant required for the mission while accounting for the amount of propellant lost during the assembly phase, which can at times be significant. Four different options of fuel delivery were derived for analysis. The first is termed the "direct launch" scenario. The others are identified by the way in which the MTV tanks are delivered, either wet, which means partly filled, or dry, which means only inert tank structure is delivered, and the tank will be filled on orbit. For both the wet and dry launch of vehicle tanks, the auxiliary propellant can be provided from either a tanker or a depot. The options were prioritized to reflect which were most critical for comparison. Dry launch of vehicle tanks was not addressed in this study. Initial evaluations indicated that dry launch of vehicle tanks would require additional ETO delivery flights that may be unnecessary. However, later safety evaluations show a benefit for dry launch of vehicle tanks.

In the direct launch scenario, the MTV tanks are modified in either capacity or number to account for the boiloff during the remainder of the assembly phase. It is desired to keep the propellants stored on orbit for as short a duration as possible, so they are delivered last to the assembly site. The mission will be ready for departure soon after the final tank is delivered to orbit.

The tanker top off case assumes the MTV to be assembled with partly full tanks in order to minimize ETO delivery flights. The tanks are then filled to capacity, or topped off, from higher capacity propellant tankers as the final step before departure. A pictorial summary of this option is given in "Tanker Top Off Reference Mission". With this method of fuel delivery, the boiloff penalties imposed by having the large amounts of propellant on orbit for the assembly phase are avoided, while each ETO launch is still being utilized efficiently. The tankers used here are essentially "dumb" tanks, carrying the minimum requirement of structure, instrumentation, and insulation to maximize the propellant load for the given launch vehicle. For the Full Science Scenario, the tanker need carry only LH2, but the Minimum Science Scenario tanker has a dual configuration, with both LH2 and LOX. Upon arrival at the assembly area, the propellant is immediately transferred to the vehicle tanks where it resides until use.

The depot top off case also seeks to decrease the boiloff losses, but through storage in better insulated depot tanks rather than MTV vehicle tanks. The depot is assumed to be deployed during an independent operation that takes place prior to the first MTV element launch. The depot tanks, which are launched full and expended after use are maneuvered to depot proximity by an advanced upper stage. A telerobotic RMS would then grapple the tankset and berth it to the depot truss. These depot tanksets are delivered after the MTV/MEV hardware has been delivered and integrated, but before the less efficient MTV tanks are launched to orbit. After the vehicle tanks have been delivered and assembly is complete, the MTV undergoes checkout procedures at the assembly node. It will then transfer over to the depot under it's own power for topping off, then depart for Mars from that location. A summary of the depot top off option is presented in "Depot Top Off Reference Mission". It should be noted that the depot in the Full Science Scenario would need to accommodate not only LH2 for the NTR, but the LOX needed for the LTV rendezvous and propellant delivery mission.

All of these fueling options, direct launch, tanker top off, and depot top off, were applied to each of the Mars exploration scenarios. The whole structure of this trade tree is shown in the chart "Depot Need Assessment Trade Tree". The dry launch options for fuel delivery were not considered, but are shown in the figure for completeness.

4.3 MANIFEST ANALYSIS

A systematic approach to the analysis of each of the trade study cases was adopted and will be summarized here. "Case Analysis Approach" shows this approach. The first step was to identify the individual components of the MEV and MTV, either CAP or NTR. Each element was characterized in terms of its mass and packaged dimensions. These discrete elements were then manifested in the launch vehicle in a way that minimized the number of ETO flights for delivery. The assumptions under which this is done are spelled out in "ETO Manifesting Groundrules and Assumptions". A complete list of vehicle components and manifests for each HLLV flight are listed in the charts "CAP Component Data" and "NTR Component Data". When considering the manifesting of the wet MTV tanks, they were filled to whatever capacity was required to bring down the ETO flight requirement. Based on this vehicle manifest, detailing the amount and delivery schedule of the MTV tanks, the boiloff and transfer losses could be calculated. This was done on the basis of a flat percentage per month for boiloff and a fixed percent rate lost per metric ton transferred. A table of tankage data used is presented in the chart "Propellant Tankage Data". These three steps, component identification, manifesting, and loss calculation, were common to each analysis. What was done with this data, however, varied from case to case, as described below.

For the direct launch case, where the lost propellant must be delivered in the MTV vehicle tanks, the losses were used to redistribute the propellant delivery by either resizing MTV tanks or adding more of them. These new additions were then worked into the manifest as shown by the directional arrow.

In the case of tanker top off, the boiloff and transfer losses were used to find the number of propellant tanker missions required to top off the vehicle for departure. These tanker missions were then added to the vehicle delivery manifest as the final mission before departure.

For the depot top off case, the propellant lost through boiloff or transfer had to be replaced with fuel stored at the depot. For both the Minimum Science Scenario and Full Science Scenario, the depot was sized to accommodate the largest propellant load required for any one Mars mission. Once the depot is sized, the ETO flights that are necessary to restock the depot are included in the ETO manifest before the MTV tanks are delivered.

4.4 INTEGRATION AND MANIFESTING RESULTS

An important indicator of the practicality and cost of a LEO system is the number of launch vehicle flights required to place and maintain the system. The charts "Minimum Science Scenario ETO Requirements" and "Full Science Scenario ETO Requirements" show these requirements for each of the fuel delivery options.

The height of each bar shows the total number of flights per year, and the shading of each portion of the bar indicates the specific type of payload carried. The order of delivery in each year proceeds from the bottom of the bar to the top, and the numbers in each boxed division show which mission the payload is to support. The order in which these items is important because it impacts directly the amount of boiloff from delivered propellants. The tanks with the largest boiloff rate or largest propellant load are placed as close to the end of the manifest as possible.

It should be noted that the propellant tanks for the CAP vehicle needed to resized in the direct launch case. The TMI tanks were taken to the capacity of the launch vehicle, 120 metric tons, while maintaining the same tank mass fraction. With an upgraded capacity of 111 metric tons of propellant, boiloff during assembly could be accommodated without additional tanks. The MOC/TEI tanks were increased to 75 metric tons for the same reason.

To summarize the Minimum Science cases, the direct launch and the tanker top off scenarios both require 11 HLLV flights for mission delivery. The depot top off case requires a total of 12 HLLV flights per mission. This is not surprising, considering the depot boiloff rate is only 0.2% per month lower than the MTV tanks. If the difference were greater, we may expect a greater benefit from using the depot.

The results for the Full Science Scenario are presented in a similar format in "Full Science Scenario ETO Requirements". A dramatic increase in the total number of flights is required by the larger number of missions. Since some of these missions fall on successive conjunction opportunities, pushing the ETO flight over the 4 per year that can be supported by one launch pad with a 90 day turnaround. If launch facilities can be prepared no faster than 90 days, these results could serve to justify the existence of at least two pads with associated equipment to support up to six launches per year. A two pad scenario would seem a prudent alternative when considering the possibility of catastrophic failure or surge operations. Even though two pads were necessary in scheduling these flights, efforts were made to spread the flights equitably, minimizing the flights per year.

In considering the direct launch option, the NTR TMI tanks had to be decreased to fit on a 120 metric ton launch vehicle completely full. Maintaining the same tanks mass fraction, these tanks were reduced to 103 metric tons each, and three to four were required, rather than two. The aft tank was increased to 75 metric tons from its previous capacity of 60 to allow for boiloff. Even though all the propellant is being launched in wet MTV tanks, some tanker missions are still required to refill the aft tank before reuse.

To summarize the resulting flight rate for the Full Science Scenario, the depot option requires one additional ETO flight over the 17 year mission model than the tanker option, and two flight more than the direct launch option. This is a significant improvement over the depot performance in the Minimum Science Scenario, where the penalty was one flight per Mars mission. This improvement is due largely to the fact that the difference between the depot and MTV tank boiloff for hydrogen only is 1.0% per month, larger than the difference for the combined LH2-LOX tanks.

Noticing this trend, a short sensitivity trade was undertaken to further investigate this relationship. it was desired to chart the behavior of the system as the MTV tank boiloff rate was increased to two times, then three times the reference value. In order to fully account for life cycle costs, the entire Mars exploration scenario had to be considered. Because it is less intensive, the Minimum Science Scenario was the first undertaken; due to time constraints, only that scenario could be completed. Only the tanker top off and depot top off fueling options were considered, because the direct option may have necessitated drastic redesign of the propellant tanks, which would not only impact the MTV design, but mission performance. Graphical results of this analysis are provided in "ETO Flight Rate Sensitivity to MTV Boiloff". Over the complete mission model, increasing the MTV boiloff by a factor of three will add another 12 flights to the tanker case, but only three to the depot case. This result is dramatic, but an even more exaggerated effect may be observed in the Full Science Scenario.

SECTION 4

MISSION INTEGRATION AND MANIFESTING

Summary Of Exploration Scenarios

Here are brief summaries of the two exploration scenarios presented for comparison. The two cases vary in a number of areas, but in general the minimum science scenario is more conservative in terms of human endeavor, cost, and technology agressiveness. For example, no bases are established in the Minimum Science scenario. The Full Science case, on the other hand, has established one base and is settling another in the course of six missions.

Although both scenarios use the longer stay time conjunction class mission, the measure of human presence weighs more heavily in the full science scenario. The crew size for the Full Science case also increases from six to twelve over the course of the program, while the crew remains at just six for the Minumum Science case.

In order to support the extensive exploration activities in the Full Science Scenario, Nuclear Thermal Rocket (NTR) technology is assumed to be mature by 2009 and utilized on the MTV, and some parts of the MTV are reusable. The Minimum Science case, although it commences six years later, is propelled by a cryogenic chemical engine, and is completely expendable.

The NTR captures Earth in an elliptical high Earth orbit and remains there until an LTV can rendezvous to retrieve the crew and transfer enough propellant for the NTR to lower itself to LEO. The CAP crew reenters in a capsule Apollo style.

Summary Of Exploration Scenarios

GENERAL DYNAMICS
Space Systems Division

MINIMUM SCIENCE SCENARIO (2015 - 2023)

- ALL CONJUNCTION CLASS MISSIONS
- AUSTERE HUMAN EXPLORATION, 3 MISSIONS WITH 6 CREW EACH
- 2 MEVs PER MISSION, 2 MANNED/TELEROBOTIC UNPRESSURIZED ROVERS
- NEARER TERM TECHNOLOGY - CRYOGENIC ALL PROPULSIVE MTV, COMPLETELY EXPENDED
- CREW REENTRY IN APOLLO STYLE CAPSULE

FULL SCIENCE SCENARIO (2009 - 2023)

- ALL CONJUNCTION CLASS MISSIONS
- MORE AGGRESSIVE EXPLORATION - 2 BASES ESTABLISHED IN 6 MISSIONS OF 6-12 CREW
- RIGOROUS SURFACE EXPLORATION UTILIZING MULTIPLE MEVs, TELEROBOTIC ROVERS
- ADVANCED TECHNOLOGY - NUCLEAR THERMAL ROCKET WITH REUSABLE ENGINE, HAB, TRUSS, AFT TANK
- ZERO-g MARS TRANSFER, LTV MUST DELIVER FUEL AND GET CREW

Tanker Top Off Reference Mission

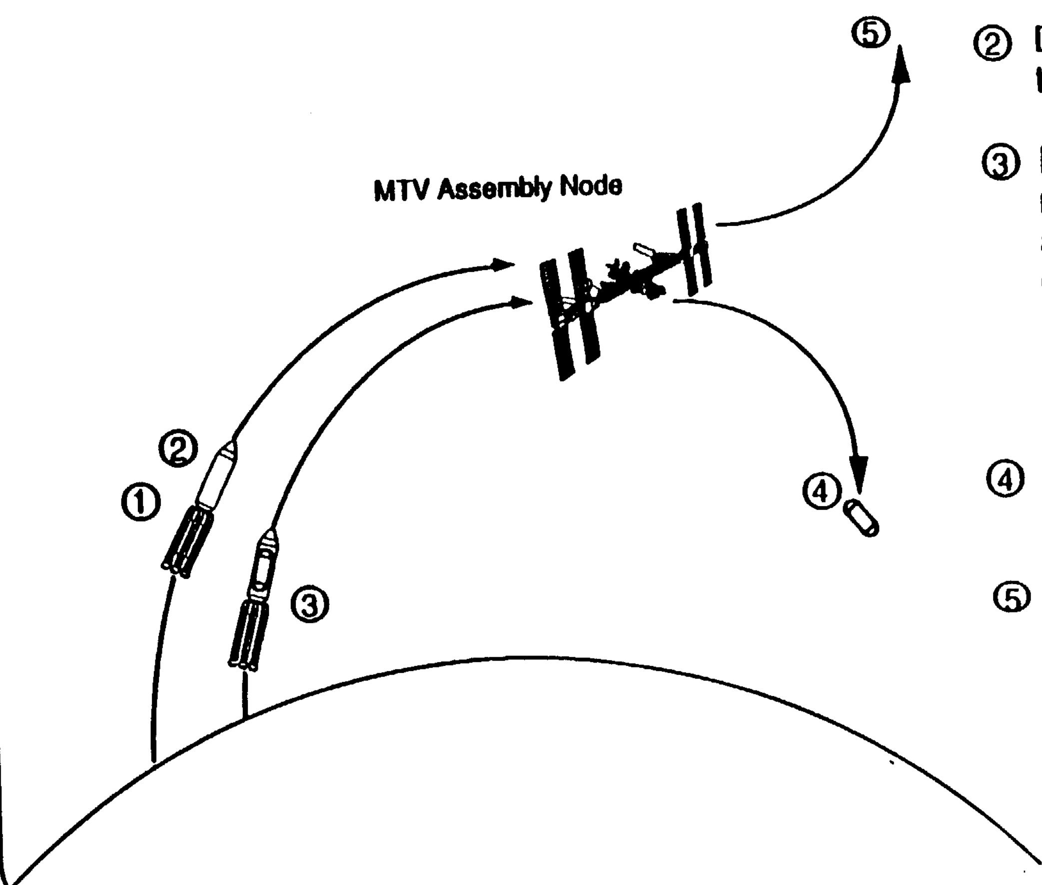
Here, the sequence of events that comprise the tanker top off scenario are summarized. The MTV assembly node is assumed to be in its final operational configuration. The only requirement imposed on the assembly node is that it has an RMS that is capable of grappling, docking, and maneuvering tanks filled to capacity with propellant.

The mission specific hardware are the first payloads to be launched. Once these items are on orbit and integrated, the MTV propellant tanks are delivered. For the tanker top off scenario, some of the propellant will be offloaded from these tanks to minimize the number of ETO flights to deliver the vehicle components. The offloaded propellant will be delivered in propellant tankers, which are less robust and poorly insulated, but offer a higher propellant capacity. The propellant in these tankers is transferred to the vehicle tanks immediately upon arrival to avoid boiloff losses.

The tankers are completely expendable, and are sent to a lower destruct orbit after they have been emptied.

For the direct launch scenario, only events 1 and 2 will be required. The MTV tanks in that scenario have been resized to allow loss to boiloff and still maintain the propellant required for the mission when it comes time to depart.

Tanker Top Off Reference Mission



① Delivery of MTV hardware to assembly node.

② Delivery of partially full MTV propellant tanks until assembly complete.

③ Propellant tankers delivered to assembly node orbit. Rendezvous performed autonomously by advanced upper stage, docking done telerobotically. Tank remains berthed for RMS retrieval or fluid line hook up.

④ Each tanker sent on destruct re-entry orbit after use.

⑤ MTV ready for departure.

Depot Top Off Reference Mission

A summary of events for the depot top off scenario is presented. The core elements of the Cryogenic Propellant Depot are to be delivered prior to the first Mars mission. The largest of the depot designs has a mass of less than 15 metric tons, and could easily be lifted by a vehicle smaller than the 120 metric ton HLLV. It is assumed that a manned crew will be required to configure the depot, so assembly, checkout, and deployment could be done at SSF. Once deployed, however, the depot is intended to be independent, requiring manned maintenance only as a contingency.

After deployment of the depot, Mars mission hardware elements will be launched to the MTV assembly node and integrated. As in the tanker scenario, the MTV propellant tanks will be launched with some propellant offloaded. Rather than following the propellant tanks to orbit in tankers, the offloaded propellant will precede the MTV propellant tanks in depot tanks. They will be stored in these heavily insulated tanks at the depot location.

Once all the depot tanks are in place, propellant tank launches to the MTV assembly node commence. When these have all been delivered and integrated to the vehicle, assembly is complete. The MTV goes through final checkout procedures at the assembly node, then transfers under its own propulsion to the depot for propellant top off. It departs for Mars from that location.

After top off, the MTV departs for Mars and the spent depot tanksets are sent on a re-entry orbit for disposal.

Depot Top Off Reference Mission

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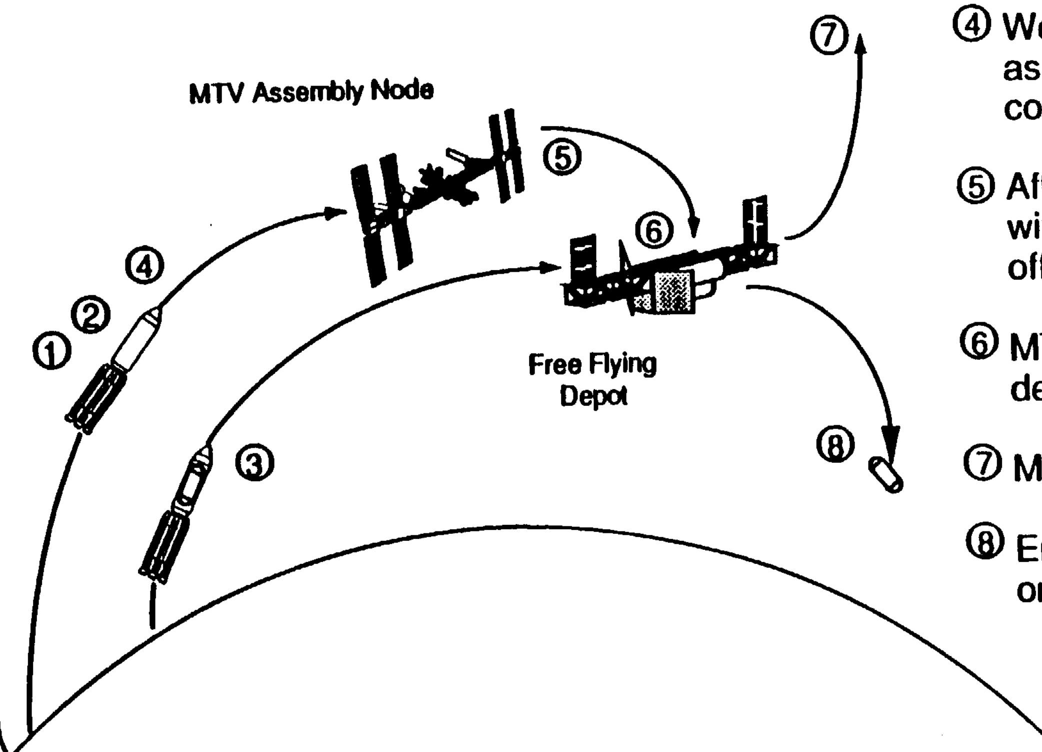


1. Delivery, assembly, and checkout of depot truss, RMS, shield, and systems
2. Delivery of MTV hardware to assembly node
3. Full depot tanks delivered to depot. Autonomous rendezvous and telerobotic docking with depot truss.
4. Wet MTV propellant tanks launched to assembly node until MTV assembly complete.
5. After checkout at assembly node, the MTV, with crew on board, transfers to depot for top off.
6. MTV propellant tanks filled to capacity at depot
7. MTV departure
8. Empty depot tanks sent on destruct re-entry orbit

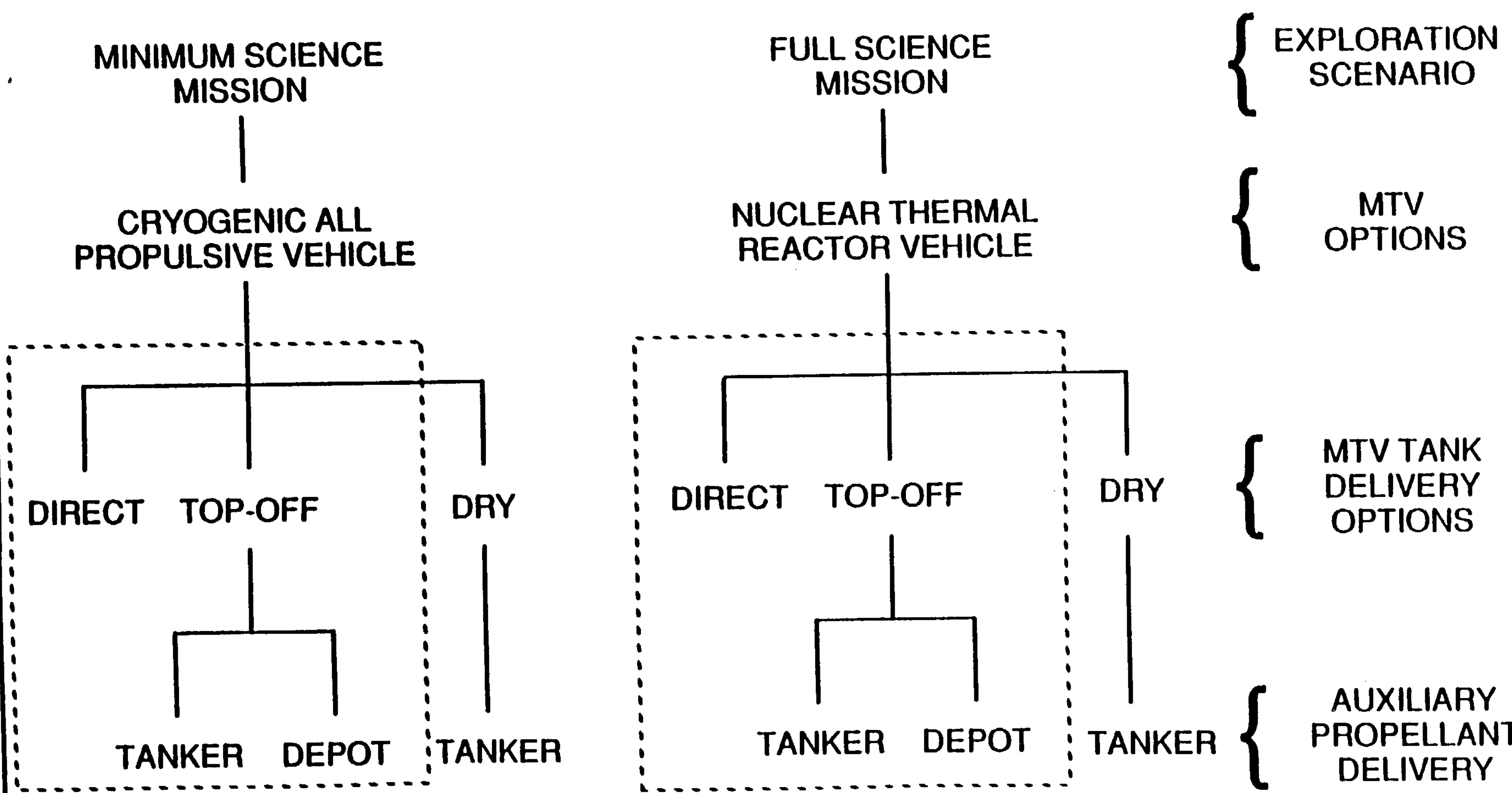
VG4-03

Depot Need Assessment Trade Tree

This chart shows the trade tree that was constructed to guide the analysis of the fuel delivery options for each of the Mars mission models. The boxes enclose those options that were addressed in the trade analysis. The volumetric inefficiency of launching completely empty MTV tanks makes dry launch a poor option, so it was not considered. However, if proximity of cryogenic propellants is the driving factor in the use of a depot, dry launch of vehicle tanks offers the safest storage option during vehicle assembly. The goal of the analysis here, though, was to derive a preliminary cost comparison between the most competitive scenarios with and without a depot.

Depot Need Assessment Trade Tree

GENERAL DYNAMICS
Space Systems Division



VG4-04

Case Analysis Approach

The method of analysis for each of the three fueling options is shown. The first step was to identify the individual components of the MEV and MTV (whether CAP or NTR) and characterize each component in terms of its mass and packaged dimensions. These discrete elements were then manifested in the launch vehicle in a fashion that minimized the number of flights for delivery. Depending on the scenario considered, the MTV propellant tanks were delivered to varying degrees of capacity. The total losses through boiloff and transfer incurred prior to vehicle departure were calculated on the basis of a flat percent rate per month for boiloff and a flat percent rate lost per transfer.

For the direct launch case, MTV tanks were resized, if necessary, so that an adequate amount of propellant remained at departure without any top off. The original tank mass fraction was maintained during all resizing. Once the number and size of the MTV tanks was resolved, these were included in the manifesting.

In the case of tanker top off, the boiloff and transfer losses were used to find the number of propellant tanker missions required to top off the vehicle for departure. These tanker missions were then added to the vehicle delivery manifest as the final missions before departure.

When considering the depot case, the boiloff and transfer losses had to be replaced with fuel that had been delivered to the Cryogenic Propellant Depot. Once the depot is sized, the ETO flights that are required to assemble and restock the depot are folded into the ETO manifest.

Case Analysis Approach

GENERAL DYNAMICS
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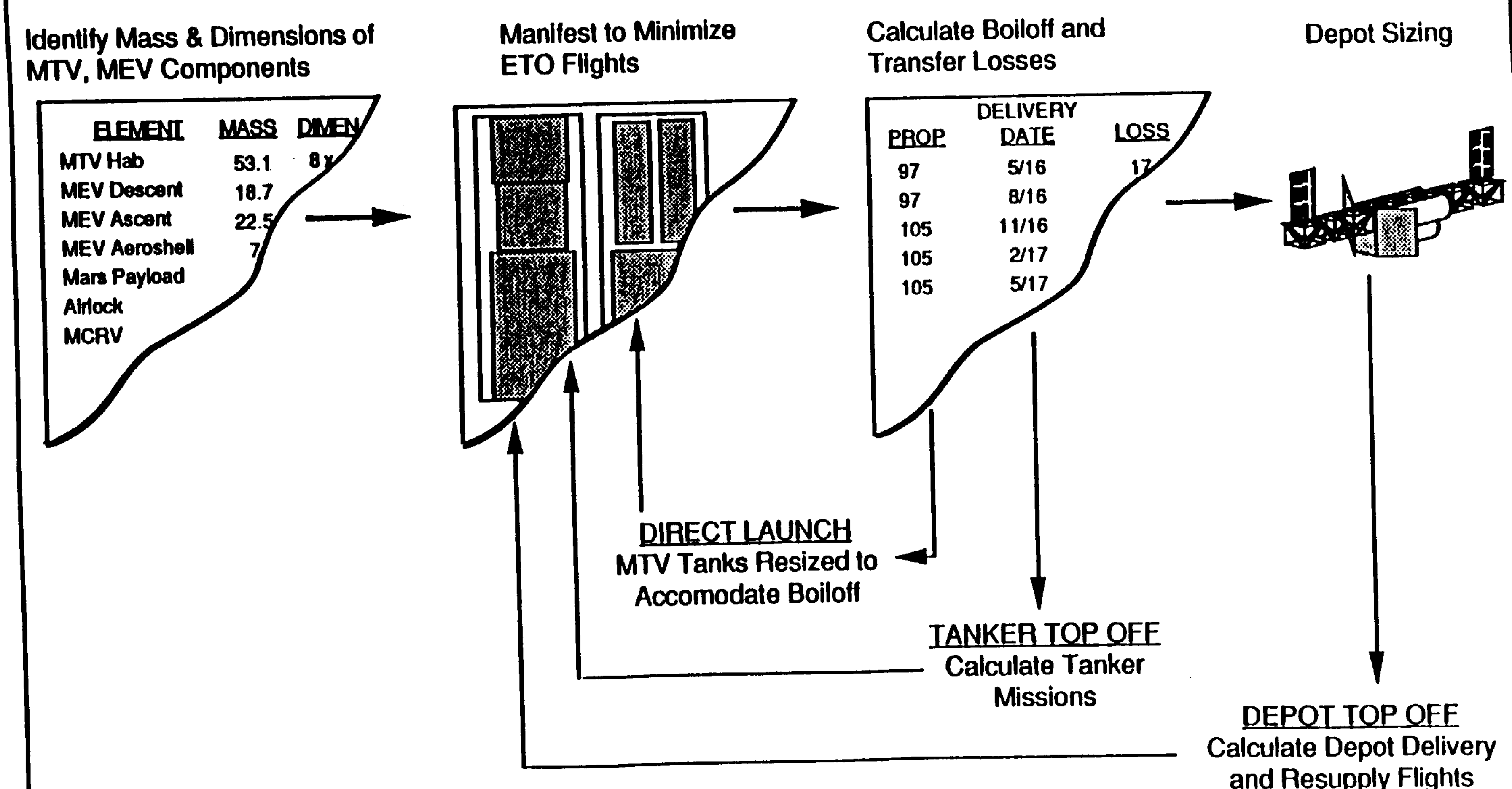

ETO Manifesting Groundrules & Assumptions

The fact that the propellant tanks of the MTV are launched at varying degrees of capacity puts a new twist into the manner that these same vehicles have been manifested in the past. Many of the groundrules and assumptions that were adopted in this analysis, which are listed above, have been employed in the past by both Boeing and GDSS.

The Boeing concept of external MEV aeroshell attachment, "Ninja turtle" style, was groundruled. This was acknowledged to degrade the in line launch vehicle performance, and 20 metric tons was groundruled to be very conservative.

The frequency of launches is constrained by the ability to turnaround facilities at the launch site. The figure of 90 days was considered to be the nominal value. Launch scheduling difficulties in the Full Science Scenario, however, necessitated the violation of this groundrule. In cases where Mars missions could not be prepared using the 90 day figure, launches were distributed as equitably as possible for the given circumstances.

The mass characteristics of even the largest of the depot designs do not merit a flight on the groundruled HLLV. With a mass of less than 15 metric tons, the depot could be delivered to SSF or the MTV node for deployment aboard a smaller vehicle or comanifested with another payload.

ETO Manifesting Groundrules & Assumptions

MTV RELATED MANIFESTING

- LAUNCH VEHICLE CAPABILITY OF 120 MT TO CIRCULAR ORBIT
- 10 x 30 SHROUD, WITH SOME NOSE CONE VOLUME AVAILABLE (AT A 15° CONE ANGLE)
- "NINJA TURTLE" AEROSHELL DELIVERY, BUT PERFORMANCE DEGRADED BY 20 MT
- CG OF PAYLOAD SHROUD CONSTRAINED AFT
- COMPONENTS DELIVERED IN LOGICAL ORBITAL ASSEMBLY SEQUENCE
- 90 DAY LAUNCH FREQUENCY ADHERED TO, WHERE POSSIBLE

DEPOT RELATED MANIFESTING

- DEPOT TO BE COMPLETELY DEPLOYED BEFORE FIRST MTV ELEMENT LAUNCH
- DEPOT CORE SYSTEMS TO BE DELIVERED BY SMALLER VEHICLE TO SSF FOR DEPLOYMENT
- LOX AND LH TANKS INTEGRATED ON GROUND AND LAUNCHED WET
- TELEROBOTIC RMS ON DEPOT AND MTV ASSEMBLY NODE CAN HANDLE MASS OF WET TANK

CAP Component Data

This chart lists the quanitity, mass, and the dimensions of the components used for the manifesting of the CAP vehicle in the trade study analyses of the Minimum Science Scenario.

CAP Component Data

ELEMENT	# / VEHICLE	DRY MASS (MT)	PROPELLANT MASS (MT)	DIMENSIONS (m)
MTV Hab	1	60.2		8 x 10
MEV Descent	2	18.7		9.5 x 20 x 4
MEV Ascent	2	22.5		9.5 x 9.5 x 5.5
MEV Aeroshell	2	7		
Mars Payload	2	25		4.4 x 13
Airlock	1	1		3 x 3
MCRV	1	7		3 x 3
TMIS Tank	5	7	97	8.4 x 15
TMIS Core	1	18.2	97	10 x 15
MOC/TEI Tank	1	5.5	56	7.6 x 15
MOC TEI Core	1	9.4	56	7.6 x 15

NTR Component Data

This chart lists the quantity, mass, and dimensions of the components used for the manifesting of the NTR vehicle in the trade analyses done on the Full Science Scenario.

NTR Component Data

ELEMENT	# / VEHICLE	DRY MASS (MT)	PROPELLANT MASS (MT)	DIMENSIONS (m)
MTV Hab	1	53.1		8 x 16
MEV Descent	2-3	18.7		9.5 x 20 x 4
MEV Ascent	2-3	22.5		9.5 x 9.5 x 5.5
MEV Aeroshell	2-3	7		
Mars Payload	2-3	25		4.4 x 13
Airlock	1	1		3 x 3
MCRV	1	7		3 x 3
Reactor,Engine	1	17		7 x 10
Shield Structure	1	2.4		7 x 7
Main Truss	1	5.3		7 x 7 x 7
Aft Tank Structure	1	1		3 x 11
TMI Tank	2	20	140	10 x 30
Aft Tank	1	9.8	60	10 x 19
Hab for 12 Crew	1	100		8 x 27
Mini MEV	2	48		10 x 13
Mini MEV Aeroshell	2	6		
Upgraded TMI Tanks	2	20	185	10 x 30
Replacement Reactor	1	10		7 x 10

Propellant Tankage Data

This chart lists the critical characteristics of the various propellant tanks used during the course of this study.

Propellant Tankage Data

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	CAPACITY (MT)	DIMENSIONS (m)	TANK FRACTION	% BOILOFF/MO.	% LOSS/TRANSFER
MTVs					
CAP					
TMI TANKS	97	8.4 x 15	.07	1.1	
MOC TANKS	56	7.6 x 15	.10	1.1	
NTR					
TMI TANKS	140/185	10 x 30	.14	1.3	
AFT TANK	60	10 x 19	.14	1.3	
TANKERs					
LOX AND LH	113	5.6 x 19	.05		4.6
LH	52	9.5 x 8.2	.13		11
	104	9.5 x 18.8	.14		11
DEPOT					
LOX AND LH	91	5.6 x 17	.16	.9	4.6
LH	45	9.5 x 11.9	.20	.3	11

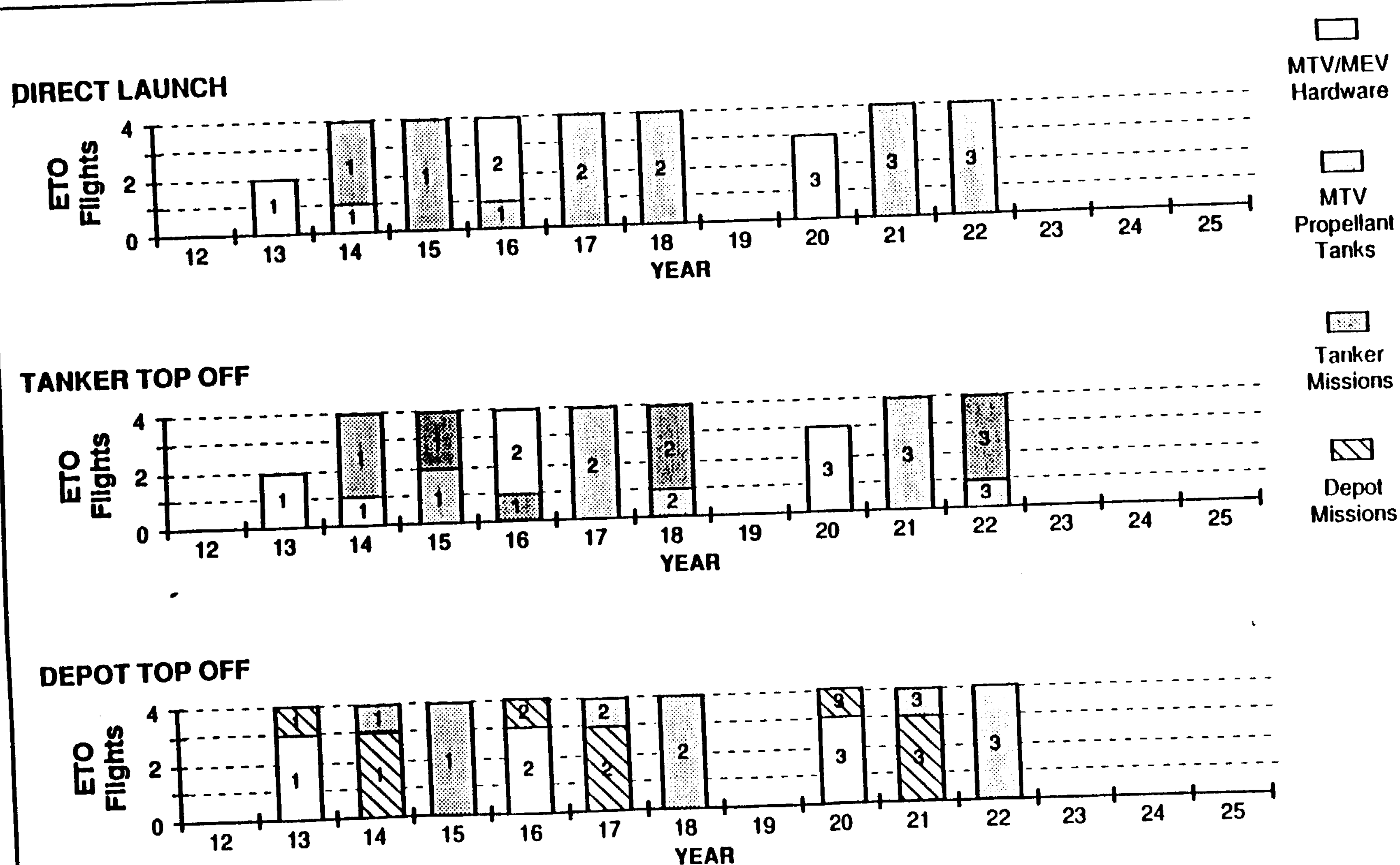
Minimum Science Scenario ETO Requirements

The three bar charts presented here enumerate the ETO flights required of a HLLV in order to fulfill each fuel delivery option under the Minimum Science Scenario. The height of each bar indicates the total number of flights per year, according to the scale at the left. The shading of each portion of the bars indicate the specific type of payload carried, as described in the legend. The sequence of delivery in each year proceeds from the bottom of the bar to the top, and the numbers in each boxed division describe which mission in the Full Science Scenario the payload is attributed to.

To illustrate, consider the option with fuel delivered via top off from the propellant tanker. The first two flights occur in 2013, carrying hardware for mission 1 (departing in Feb. 2016) to the assembly node. The first flight to take place in 2014 carries the same type of payload destined to be used for the same Mars mission. The remaining three flights in 2014 deliver the partially filled MTV propellant tanks. The propellant tanker missions commence in the latter half of 2015, and continue unitl departure for Mars after the first ETO flight in 2016. The next three flights in 2015 then begin the delivery of mission hardware for mission 2.

The bottom line of this analysis is to show that the inclusion of a Cryogenic Propellant Depot to the Minimum Science Scenario will require an additional HLLV flight per Mars mission. This result is not surprising considering the boiloff rate for the MTV vehicle tanks is just 0.2% per month greater than the depot tanks. If the difference were greater, the depot would offer much more efficient storage of propellant, and a cost advantage.

Minimum Science Scenario ETO Requirements



VG4-10

Full Science Scenario ETO Requirements

The bar graphs on this chart follow the same format introduced on the previous chart, but it may be more difficult to track ETO flights allocated to a specific Mars mission because there are three more Mars missions in the Full Science Scenario, most falling on successive conjunction mission opportunities.

This increased flight rate, among other factors, contributes to the violation of the 90 day launch frequency constraint imposed by the ability to turnaround ground based launch facilties. In all three propellant delivery options, this constraint must be infringed in order to depart for Mars at the specified date.

As expected, the depot is becoming a more efficient option as the difference between MTV tank boiloff to depot tank boiloff increases. In the Full Science Scenario, the depot option adds one additional ETO flight to the tanker option, and just two flights to the direct launch option over a 17 year Mars mission model. That is quite an improvement over the one launch per Mars mission penalty the depot imposed on the Minimum Science Scenario.

Bottom line: The use of a Cryogenic Propellant Depot looks significantly better in a an advanced architecture, where there are many Mars missions and the high thrust propellant of choice is highly volatile liquid hydrogen, which the depot can store more efficiently than light weight vehicle tanks.

Full Science Scenario ETO Requirements

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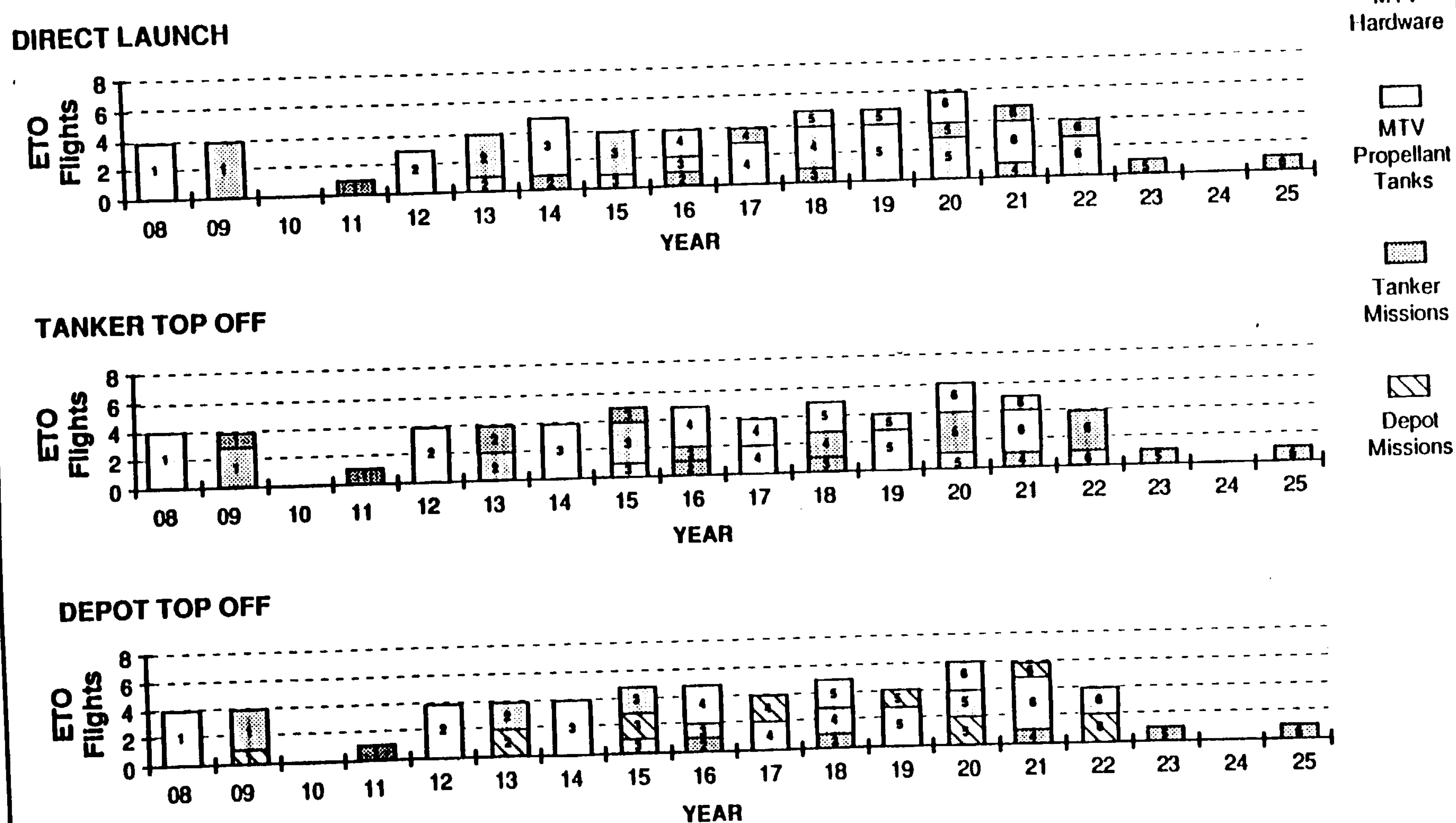

ETO Flight Rate Sensitivity To MTV Boiloff

The three bar charts presented here show the difference obtained in the required ETO flight rate as the CAP vehicle tank boiloff rate was doubled, then tripled. Only the tanker and depot methods of propellant delivery were considered for this trade study. The reason for omitting the direct launch scenario is that increasing the boiloff above the original rate would have entailed redesign of the tanks to accomodate the added propellant, the impacts of such a redesign were beyond the scope of this study.

It is evident that over the complete Mars mission model, increasing the MTV boiloff rate by a factor of three will add another 12 flights to the tanker case, but only 3 to the case where a depot is used.

ETO Flight Rate Sensitivity To MTV Boiloff

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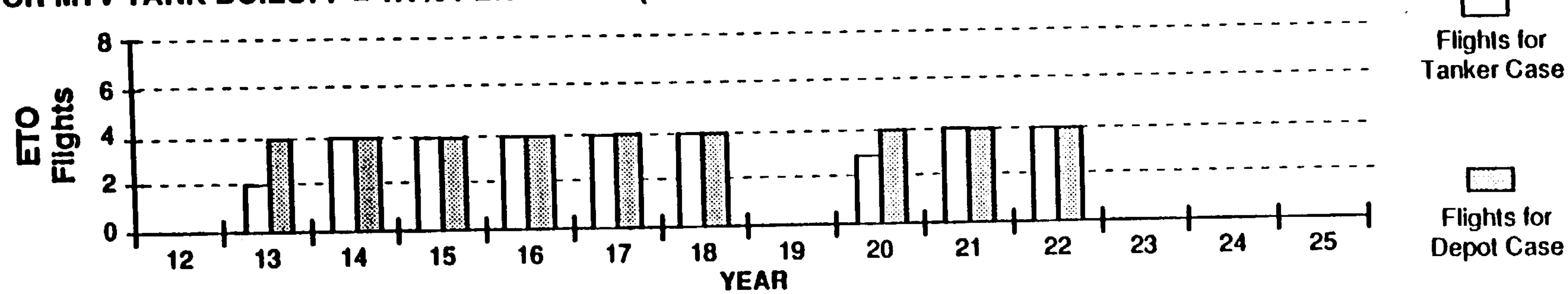
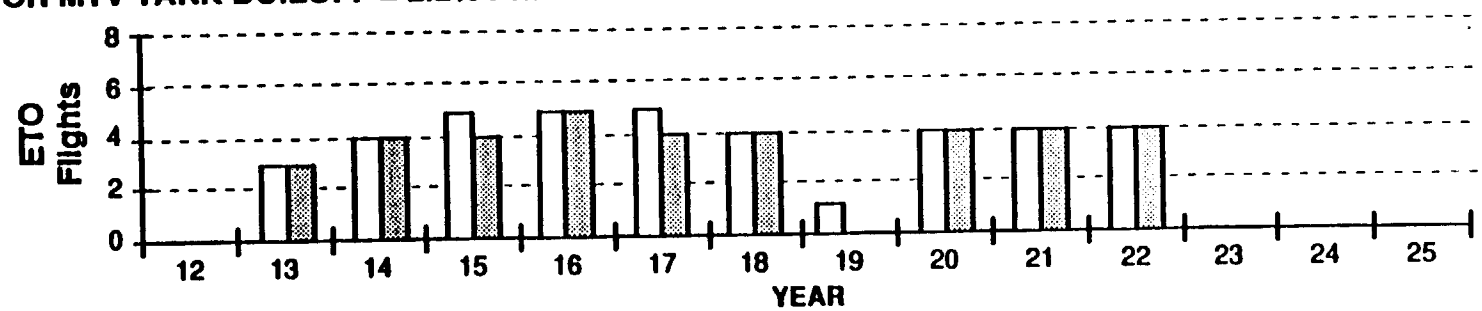
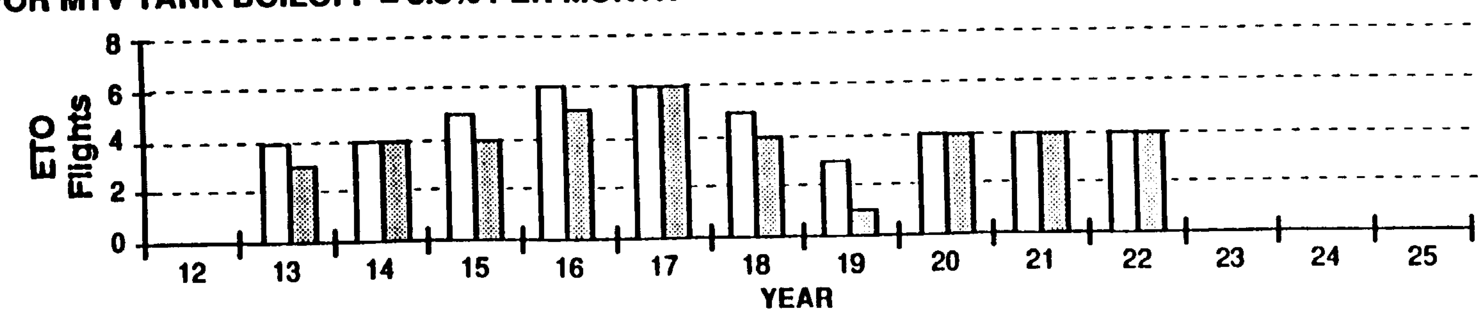
FOR MTV TANK BOILOFF = 1.1% PER MONTH (ORIGINAL VALUE)



FOR MTV TANK BOILOFF = 2.2% PER MONTH



FOR MTV TANK BOILOFF = 3.3% PER MONTH



SECTION 5

PROPELLANT DEPOT CONCEPTS

5.1 LEO PROPELLANT DEPOT CONCEPTS

5.1.1 Concept Description

The depot concepts presented here reflect the current understanding of all-passive thermal controlled, zero-g fluid transfer depot technology, which was initially pursued in the Long Term Cryogenic Storage Facility (LTCSF) report issued by General Dynamics Space Systems in October 1988. Currently work has been conducted on zero-g and artificial gravity transfer of liquid propellants which has been included within this report in a detailed section.

Four LEO orbital propellant depot configurations were identified as potential candidates and are shown in the charts, "Depot Concept for Support of CAP Vehicle", "Depot Concept for Support of NTR Vehicle", "ET Shielded Depot Configuration", and "Gravity Gradient Stabilized Depot Configuration".

Both CAP and NTR supporting depot configurations utilize a composite truss modeled on the current Warren type (alternating battens) truss baselined for use on SSF due to its improved torsional stability and single member failure tolerance. It will be of a collapsible design to facilitate stowage and deployment in order to minimize EVA requirements. The keel section of the truss in the NTR configuration offers attach points for tank hardpoint fixtures, yet offers little in the way of strengthening, such that the members for both configurations would be sized similarly by flexibility and dynamics constraints.

Both concepts include all-passive, vented tanks. The CAP configuration makes use of three 200,000 lb LH2/LOX tanks proscribed in the LTCSF study. The largest tank size was selected based on payload lift capability. The NTR configuration makes use of nine all hydrogen tanks and one oxygen tank to supply the MTV with hydrogen propellant as well as an LTV and the depot's own reboost thrusters with both hydrogen and oxygen for their bi-propellant thrusters. Both configurations employ Vapor Cooled Shields (VCS's) interspersed amongst MLI blankets for passive cooling, utilizing boiloff from the hydrogen tanks as the heat transfer fluid within the VCS. Both are zero-g transfer concepts which make use of a Liquid Acquisition Device (LAD).

The tanks are shielded by a separate, deployable "winged" shield comprised of aluminum sheet, MLI, and a stand off to protect against both micrometeoroid and orbital debris damage. Making the shield out of a single sheet of aluminum is prohibitively weight consuming. By employing a standoff in the design, ejecta from the less weighty outer aluminum barrier are redirected at more incident angles to the inner MLI layer and over a larger area to minimize energy concentration of the initial impact. The MLI blankets then absorb the remainder of any energy imparted by projectiles penetrating the outer shield, thereby leaving the tank wall free from the shielding system. The shielding is maintained separately from the tanks to minimize cross-sectional area and weight as well and to avoid the recurring weight penalty involved with launching the shield as a portion of the tank. The winged configuration is employed to shield the faces of the depot most vulnerable to impact by orbital debris as illustrated in the chart "Shielding Considerations". The 45° angled wings attempt to cover the tanks from debris while minimizing surface area, and thus weight.

Propellant requirements for the depots were summarized and shown in the charts, "CAP Depot Configuration Propellant Requirements" and "NTR Depot Configuration Propellant Requirements".

Tanks were sized according to manifested propellant needs per vehicle mission. Some missions require that MTV tanks be launched partially filled such that the launch vehicle payload envelope is more fully utilized, tanks being primarily mass rather than volume constrained. This "off-loaded propellant" is then accounted for in depot capacity. Additionally, top off propellant is used for fuel which is off-loaded for mass constraint reasons. Boiloff and transfer losses were also calculated for the fluid based on their launch sequence and duration on orbit. These components sum to the total capacity a depot must handle in order to supply each vehicle with its full complement of propellant prior to launch. The totals which were used to size both depot configurations are presented in the charts "CAP Depot Configuration Propellant Requirements" and "NTR Depot Configuration Propellant Requirements". It can be seen that an increase in propellant need occurs in Mission 5-6 for the NTR vehicle. The NTR depot configuration reflects this capacity. However, for the first four missions, propellant requirements are such that only seven hydrogen tanks need be supplied.

Solar power is generated by the deployable panels located at the ends of each truss. The panels were sized based on a 400 W/tank need for the LH2 tanks and the 660 W/tank need demonstrated by the LH2/LOX tanks as explained in the chart "Solar Panel/Power Parameters". Both employ alpha and beta joints for sun tracking. GaAs cells are baselined for improved efficiency and robustness. On the CAP configuration, solar panels are located on the space side of the depot to minimize shadowing by the tank shielding. The NTR configuration utilizes larger solar panels both for the increased power demand represented by the greater number of tanks, and because of the increased shadowing which occurs due to the large shield employed in the configuration.

Reboost capability has been baselined as bi-propellant 100 lb thrusters which feed directly from the depot's own stores, alleviating the need for large quantities of monopropellant and special dedicated resupply missions. RCS is handled by two sets of 870 lb hydrazine thruster pods. GN & C is handled autonomously on board and is monitored at station through telemetry.

5.1.2 Operational/Integration Issues

In order to better utilize the mass and volume capacity of the launch vehicle, it was originally believed that mixing components from the MTV and depot would facilitate manifesting. However, it was found that sufficient flexibility existed in the extensive MTV component list to allow either full volume or mass capacity utilization without the need to mix components from the depot. Furthermore, no immediate gain could be realized by separating the LH2 and LOX tanks which comprise the propellant tanks for the CAP Configuration depot.

In order to avoid impacting the MTV assembly schedule or assembly crew load, the depot launch/ assembly should occur before inception of the MTV launch/assembly phase. Both assembly phases would benefit from an initial platform (either STS or SSF) from which to deploy the truss structure. In order to facilitate this deployment, the truss structure shall incorporate a deployable design to minimize EVA demands.

Telemetry for onboard equipment must be incorporated to allow man-tended, remote monitoring of the depot systems, as no manned habitat or crew safe haven will be available at the depot. Life support will be furnished by the crew transfer vehicle only.

RMS is required for transfer of the cryogenic storage tanks from the LTV or other ferrying system to the depot upon orbital insertion.

Reboost/deboost operations will be monitored remotely. These engines will feed from tank residuals onboard the depot, necessitating resupply missions be timed to coincide with the reboost phase.

These issues are summarized in the chart "Operational/Integration Issues.

5.2 MARS PROPELLANT DEPOT ASSESSMENT

5.2.1 Mars Depot Architecture Concept Options

Many Mars system depot architecture options are capable of supporting a large, long-term Mars exploration program. Several are shown in "Mars Depot Architecture Concept Options" along with their potential propellant delivery routes (arrows).

Wherever primary propellant production sites are located - Phobos, Deimos, or Mars surface - a surface propellant depot capable of storing and protecting cryogens against the ambient environmental conditions will be required. At least two orbital locations immediately suggest themselves: low Mars orbit (LMO) and high Mars orbit (HMO); an HMO depot would have orbital altitudes comparable to the moons. An LMO depot would be a logical alternative if propellants were originating on Mars, although it could also be supported from Phobos and Deimos. Likewise, propellants originating on the moons would suggest the use of an HMO facility. The most attractive option will be determined by specific Mars mission and program scenarios.

5.2.2 Mars Environmental Issues

Operating near and on Mars means encountering an entirely new set of orbital, thermal, gravitational, atmospheric, and surface environments that can be crucial to the operation of a Mars orbital propellant depot. Summarized in "Mars Environmental Issues", some of these parameters are well-known but others have not yet been well-specified.

At 1.5 AUs from the Sun, Mars intercepts less than half of Earth's solar flux and this value varies considerably due to Mars' orbital eccentricity (solar flux at Mars is 490 to 710 W/m2). The average Bond albedo (0.16) results in the planet having much lower daytime and nighttime blackbody surface temperatures than Earth, so objects in Mars orbit receive less thermal radiation from Mars than they would comparable orbits around Earth. We have estimated that one of our LEO depots would experience a 25% decrease in boiloff rate when in a similar Mars orbit (it's also a function depot spatial orientation).

The general meteoroid environment near Mars is well-known from several spacecraft missions over the last three decades. However, there are good theoretical reasons to suspect that belts of dusts may exist in orbits near the satellite moons (particularly Phobos). These may pose a threat to the integrity of any spacecraft or human exposed to them for too long. Plans for Mars depots must anticipate and evaluate this potential threat to long-term, routine operations in these orbits.

The Martian planetary gravity field has sizeable irregularities that are well characterized. However, if operations are contemplated near Phobos and Deimos, their complex local gravity field must be better understood. We show in "Mars Environmental Issues" one theoretical solution for the escape velocities as a function of location and direction on Phobos. They range from 3.5 m/s toward Mars at the sub-Mars point to 15.5 m/s 90 degrees away at the north pole. Deimos' gravity variations are smaller in amplitude.

Operating on the Mars surface itself is a complex, potentially dangerous, highly challenging proposition. The 24 hour diurnal cycle may drive the use of nuclear power on the surface. Diurnal and seasonal variations in surface temperature and atmospheric properties will require careful monitoring of cryogenic fluids being produced or stored near the surface.

5.2.3 Mars Depot Location Options

Five general locations (and 9 more specific regions) suggest themselves as potential locations for Mars propellant depots; they are: Mars orbit (free space), Phobos, Deimos, Mars' surface, and the Sun-Mars libration (L) points. A preliminary list of advantages and drawbacks is shown for each region in "Mars Depot Location Option Summary". This Mars depot study is not intended to exhaustively assess the attributes of each location, but merely to present preliminary observations and suggest fruitful avenues for future analysis.

5.2.3.1 Mars Orbits. Each location is typically characterized by its environmental attributes, the nearest location of primary propellant production, and its relation to the anticipated Mars system infrastructure and exploration program. For example, three types of Mars orbits appear to be potentially useful as sites for Mars propellant depots: 1) high, elliptic and inclined orbits, 2) high, circular, equatorial (HMO, high Mars orbit) orbits, and 3) low circular orbits (LMO, low Mars orbit). High elliptical orbits typically will have orbital elements similar to incoming interplanetary spacecraft and provide a convenient mechanism to execute plane changes. Because a depot would only experience close approaches to Mars briefly each orbit, it would suffer relatively moderate thermal loads. Nevertheless, these orbits are probably the least attractive sites for a propellant depot because of their relative inaccessibility to the Mars surface, LMO, and even the Martian moons. Because incoming MTV orbits are also quite variable, it would be difficult to find an optimal location in such an orbit that would be consistent with the anticipated Mars exploration scenarios.

High circular orbits (i.e. those with semimajor axes comparable to the moons) are accessible to Phobos and Deimos and are far removed from the Mars thermal radiation source. Vehicles operating from depots in these orbits would still face a high delta-v to Mars surface as well as some unique micrometeoroid hazards; these are due to regolith that has been propelled by impacts into the complex Mars-satellite gravity fields near the moons.

Low Mars orbits are convenient to surface operations (the ultimate focus of any Mars program) and the most likely location of primary propellant production: the Mars surface. A depot in LMO also experiences the most extreme thermal load due to Mars and must climb most of the Mars' gravity well to escape. Without an in-depth evaluation of all these factors, our preliminary suggestion is that LMO seems the most likely location for an early depot within the Mars system.

5.2.3.2 Phobos and Deimos. The Martian moons provide many strategic and operational advantages associated with the exploration of Mars. Their very weak, milli-g fields and potential for in situ propellant production make them potentially pivotal early targets for human exploration and utilization. Consistent with the most recent Earth-based spectral evidence and data from the Soviet Phobos probe, the moons of Mars are expected to possess bulk compositions characterized by significant amounts of water, hydrated silicates, and hydrocarbons. Models of asteroids (Phobos and Deimos may be captured asteroids) include the possibility that significant ground ice could also exist. Theoretical studies suggest that if ground ice ever existed in Phobos (and presumably on Deimos), it should still be there.

Thus, in addition to being an accessible potential primary propellant production site, Phobos and Deimos can also serve as thermal shadows from Mars and even provide a milli-g gravity field for fluid acquisition. Because the moons have such low albedos, the fact that their surfaces are exposed to interplanetary space does not guarantee them extremely low temperatures. Indeed, heat conduction from Phobos or Deimos themselves may neutralize any anticipated advantages of locating the depot near or on their thermally-sheltered anti-Mars sides. It is also important to realize that the low-gravity, dust-filled environments of Phobos and Deimos are very unusual and by the time humans venture forth to Mars it is unlikely that any human operations database will exist for such objects. These challenges suggest a more cautious approach to the exploitation of the Martian moons than their bare vital statistics might suggest.

Propellant depots might also be located near (but not actually on) Phobos or Deimos. Stable orbits are known to exist around these tiny objects and their very minimal local gravity field might make stationkeeping a viable option. While much of the Mars thermal shield might be forfeited, the major advantage of a near-satellite orbital location would be to avoid the complex surface terrain and environmental hazards of the moons each time a vehicle rendezvous with the depot. However, it should be noted that wherever propellant depots are located, some storage/transfer function will be required for use of the propellants in MEVs and MTVs.

5.2.3.3 The Mars Surface. The most likely site for some type of primary propellant production is on the surface of Mars using the regolith and/or the atmosphere. The 1/3 g surface gravity field should make fluid and human operations relatively normal. A Mars surface depot would not have most of the flight subsystems required of an orbital facility. However, the Martian surface environment - particularly its thermal radiation - will provide the biggest challenges for producing and storing cryogenics. Surface atmospheric conditions (including 24-hour diurnal cycles) will probably require use of nuclear systems for power, unlike the situation for orbital depots. One solution to large expenditures for power is to store the future cryogenics as water and them split/liquify them as their use requires.

For completeness, the Sun-Mars L points are included as potential depot locations. The L1 point, being over one million km from Mars, will not have a significant planetary thermal problem, and the location is conducive to interplanetary operations. Nevertheless, the site is sufficiently remote from the Mars propellant production locations and the remainder of the Mars infrastructure to call into question whether these L points possess any real potential for a major contribution to any near-term Mars exploration program.

5.2.4 LEO-Mars Depot Commonalities

5.2.4.1 Preliminary Assessment. System, technology, and operational similarities between depots planned for potential Martian use and those contemplated for LEO are important considerations. If significant commonalities exist between LEO and Mars systems, and if a mission indeed exists for a Mars depot, then cost and schedule savings might be obtained. This top-level look at commonalities suggests that, in some locations near Mars, the LEO system might be overdesigned and susceptible to significant transference. Conversely, it is also possible that the identification of both an important mission for a depot at Mars coupled with significant LEO-Mars depot commonalities, might influence planners to recommend that such a depot be built even if the case for a LEO depot is only marginal. "LEO-Mars Depot Commonalities - Preliminary Assessment" shows our initial assessment of the likely commonalities between LEO and Mars depots for the major subsystems and operations at four general Mars locations (Phobos and Deimos are considered together).

5.2.4.2 Major Subsystems. Our baseline thermal management system is all passive and vented. A similar system might be used in either LMO or HMO where the boiloff due to planetary thermal radiation will be less than in LEO. Particularly in LMO, a Martian depot might benefit from the capability to avoid or reliquefy any boiloff due to the scarcity of Martian propellants and/or because of the presence of science instruments or manned operations in the vicinity of the depot. It is unlikely such considerations will be important on the dusty, milli-g worlds of Phobos and Deimos, although the long-term storage of cryogenics on Mars itself will probably require refrigeration. Another alternative is to produce water from planetary raw materials and then store it until a short interval before the propellants are required, when the water would be split and liquefied. This would require considerable schedule knowledge and control.

With the exception of the Mars surface and its moons, all Martian orbits could offer zero-g conditions for fluid acquisition and transfer similar to the case for LEO systems. While tank changeout is envisioned for LEO, this requires a powerful RMS such as that at SSF and it is unclear whether such a system will be available near Mars. In its absence, fluid transfer would be the preferred technique. On the surfaces either technique should be possible.

Nuclear power systems seem preferable on planetary and satellite surfaces because of their diurnal cycles and atmospheric environments. If frequent occultations by Mars are a problem in LMO, its depot might also require a nuclear system. HMO is not frequently occulted by Mars.

Presently, the orbital environment of Mars does not suffer from debris hazards as such although there is a natural micrometeoroid population. In particular, the orbits of Phobos and Deimos are likely to feature enhanced dust lanes because of impacted regolith previously resident on the moons' surfaces.

As in LEO, a depot near Mars will probably require a TDRSS-like system to support it. Depots on Mars or Phobos and Deimos will have well-defined surface locations. It is also likely that depot reboost requirements in LMO will be much less than those in LEO due to Mars' relatively thin (6 mb) atmosphere.

Unlike LEO depots, it may be more efficient to locate science payloads on or near the Mars vicinity depot. As has already been indicated, this can have significant effects on depots subsystems and operations.

5.2.5 Conclusions and Issues For Further Study

"Issues For Further Study" lists a few major areas that could benefit from more study to further define the potential role for a Mars propellant depot.

The Minimum Science Program and Full Science Scenario are only two of the many scenarios that can be envisioned for human Mars exploration programs. It is unlikely that a Mars propellant depot would be required for the former, but any science, resource, or settlement scenario that anticipates significant operations near and/or on Mars will necessitate the use of in situ resources, in particular the in situ production of propellants. A depot can be a logical step in many of them. There is the suggestion from this analysis that a Mars depot should be examined further.

It is clear that several of the options for a Mars depot architecture are attractive and should be probed and evaluated. It is tempting to suggest that the Martian moons are so accessible, approachable, and wet that they should preempt all other potential targets for early human exploration and utilization. However, their unusual, potentially threatening environments may weaken many of their obvious advantages and drive us to the surface of Mars for more Earth-like working environments. This question needs much more study.

In addition to the locations of the primary propellant production sites, the Mars depot architecture is also influenced by environmental, space infrastructure, and exploration program considerations. The problem is interwoven in a complex way with all the other complicated exploration and architecture plans for Mars.

It is likely that depots in orbit around Mars could share much commonality with those in LEO. However, surface systems will be different in many ways. Before the influence of commonality can be realistically ascertained, the role of propellant depots in the Mars exploration program must be more fully defined.

It is possible that depots near Mars will be enhanced by or require new technologies. For example, higher system reliability and more autonomy would be valuable at Mars. Likewise, depots capable of interfacing with propellant production facilities on Mars or its moons must eventually be developed. If depots are located on or near Phobos or Deimos, new depot technologies will have to accommodate these environments.

5.3 EVALUATION OF ON-ORBIT SAFETY HAZARDS

The key safety hazards associated with orbital vehicles and supporting systems have been identified and provide a source of concern for on-orbit operations and possible vehicle loss with consequent Mars mission disruption. These issues are presented in "Key Potential Vehicle Safety Hazards".

An important hazard is the threat of the pressurized propellant tanks becoming inadvertently propulsive. This is possible if a meteoroid or space debris impact and punctures the vehicle tanks. A rupture due to overpressurization could have a similar effect. We do not consider ignition to be a likely possibility in space because of the need for appropriate pressurization and ignition energy. It is possible to obviate these hazards by appropriately shielding the vehicle tanks or providing a propellant depot as shown in "Depot Advantages in Safety Hazards Abatement" with adequate tank shielding and monitoring the evolution of pressure within the structures. Providing a propellant depot to fuel-up the Mars vehicles just prior to mission operation allows the vehicle tanks to be empty during most of the stay time in LEO orbit, thereby reducing the probability of occurrence of a catastrophic event.

Hazards during EVAs are potentially dangerous for the crew. This is particularly important for the case of tank leaks that physically contact the EVA suits. Accidental firing of a thruster near a crewperson must also be avoided. Avoiding leaks, monitoring EVAs, and making the times of thrusting and EVAs mutually exclusive will help.

If tank changeout is the transfer mechanism of choice then a large RMS will be a requirement, and with it the possibility of a malfunction and accident. Likewise, if we eliminate the shuffling of large tanks, large volumes of propellants will have to be pumped into empty vehicle tanks. Care must be taken to execute these sequences nominally and, when possible, minimally.

Nuclear vehicles become hazardous when in the vicinity of other systems because of their radiation fields associated with their reactors. Proximity operation rules must be established for nuclear vehicles (particularly in LEO) as well as appropriate shielding of humans and vital components from their radiations.

SECTION 5

PROPELLANT DEPOT CONCEPTS

Depot Concept for Support of CAP Vehicle

Passive thermal controlled, vented zero-g transfer depot for support of CAP vehicle.

RISK ANALYSES

Risk analyses were conducted to develop an initial risk assessment for the various architectures. This presentation of risk analysis results considers development risk, man-rating requirements, and several aspects of mission and operations risk.

Development Risk

All of the architectures and technologies investgated in this study incur some degree of development risk; none are comprised entirely of fully developed technology. Development risks are correlated directly with technological uncertainties. We identified the following principal risks:

Cryogenics - High-performance insulation systems involve a great many layers of multi-layer insulation (MLI), and one or more vapor-cooled shields. Analyses and experiments have indicated the efficacy of these, but demonstration that such insulation systems can be fabricated at light weight, capable of surviving launch g and acoustics loads, remains to be accomplished. In addition, there are issues associated with propellant transfer and zero-g gauging. These, however, can be avoided for early lunar systems by proper choice of configuration and operations, e.g. the tandem-direct system recommended elsewhere in this report. This presents the opportunity to evolve these technologies with operations of initial flight systems.

Engines - There is little risk of being able to provide some sort of cryogenic engine for lunar and Mars missions. The RL- 10 could be modified to serve with little risk; deep throttling of this engine has already been demontrated on the test stand. The risk of developing more advanced engines is also minimal. An advanced development program in this area serves mainly to reduce development cost by pioneering thecritical features prior to full-scale development.

Aerocapture and aerobraking - There are six potential functions, given here in approximate ascending order of development risk: aero descent and landing of crew capsules returning from the Moon, aerocapture to low Earth orbit of returning reusable lunar vehicles, landing of Mars excursion vehicles from Mars orbit, aero descent and landing of crew capsules returning from Mars, aerocapture to low Earth orbit of returning Mars vehicles, and aerocapture to Mars orbit of Mars excursion and Mars transfer vehicles. Figure x.x provides a qualitative development risk comparison for these six functions.

continued

Development Risk Assessment For Aerobraking By Function

MISSION FUNCTION	BRAKE SIZE	ATMOSPHERE KNOWLEDGE & UNCERTAINTY	TARGET FOR ENTRY: GN&C PRECISION	HEATING/TPS	AERO PASS GN&C PRECISION REQUIRED
Lunar return Earth landing	Small, no ass'y required	Accurate knowledge, low uncert. effect	Very high	State-of-the-Art	State-of-the-Art
Lunar return Earth landing	Moderate requires assembly	Accurate knowledge, high uncert. effect	Very high	State-of-the-Art	Believed State-of-the-Art
Mars landing from orbit	Large, requires assembly	Poor knowledge, low uncert. effect	Can be high, e.g. done from Mars orbit	State-of-the-Art	Believed State-of-the-Art
Mars return Earth landing	Small, no ass'y required	Accurate knowledge, moderate uncertainty effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Accurate knowledge, high uncert. effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Poor knowledge, high uncert. effect	Poor, unless nav-aids in Mars orbit	High heating rates, some TPS advancement needed	Advancements required

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Aerocapture of vehicles requires large aerobrakes. For these to be efficient, low mass per unit area is required, demanding efficient structures made from very high performance materials as well as efficient, low mass thermal protection materials. By comparison, the crew capsules benefit much less from high performance structures and TPS.

Launch packaging and on-orbit assembly of large aerobrakes presents a significant development risk that has not yet been solved even in a conceptual design sense. Existing concepts package poorly or are difficult to assemble or both. While the design challenge can probably be met, aerobrake assembly is a difficult design and development challenge, representing an important area of risk.

Nuclear thermal rockets - The basic technology of nuclear thermal rockets was developed and demonstrated during the 1960s and early 1970s. The development risk to reproduce this technology is minimal, except in testing as described below. Current studies are recommending advances in engine performance, both in specific impulse (higher reactor temperature) and in thrust-to-weight ratio (higher reactor power density). The risks in achieving these are modest inasmuch as performance targets can be adjusted to technology performance.

Reactor and engine tests during the 1960s jetted hot, slightly radioactive hydrogen directly into the atmosphere. Stricter environmental controls since that time prohibit discharge of nuclear engine effluent into the atmosphere. Design and development of full containment test facilities presents a greater development risk than obtaining the needed performance from nuclear reactors and engines. Full- containment facilities will be required to contain all the hydrogen effluent, presumably oxidize it to water, and remove the radioactivity.

Electric Propulsion Power Management and Thrusters - Power management and thrusters are common to any electric propulsion power source (nuclear, solar, or beamed power). Unique power management development needs for electric propulsion are (1) minimum mass and long life, (2) high power compared to space experience, i.e. megawatts instead of kilowatts, (3) fast arc suppression for protection of thrusters. Minimizing mass of power distribution leads to high distribution voltage and potential problems with plasma losses, arcing, and EMI. Thus while power management is a mature technology, the unique requirements of electric propulsion introduce a number of development risks beyond those usually experienced in space power systems.

Electric thruster technology has been under development since the beginning of the space program. Small thrusters are now operational, such as the resistance-heat-augmented hydrazine thrusters on certain communications spacecraft. Small arc and ion thrusters are nearing operational use for satellite stationkeeping.

Space transfer demands on electric propulsion performance place a premium on high power in the jet per unit mass of electric propulsion system. This in turn places a premium on thruster efficiency; power in the jet, not electrical power, propels spaceships. Space transfer electric propulsion also requires specific impulse in the range 5000 to 10,000 seconds. Only ion thrusters and magnetoplasmadynamic (MPD) arc thrusters can deliver this performance. Ion thrusters have acceptable efficiency but relatively low power per unit of ion beam emitting area. MPD thruster technology can deliver the needed Isp with high power per thruster, but has not yet reached efficiencies of interest. Circular ion thrusters have been built up to 50 cm diameter, with spherical segment ion beam grids. These can absorb on the order of 50 kWe each. A 10 MWe system would need 200 operating thrusters. The development alternatives all have significant risk: (1) Advance the state of the art of MPD thrusters to achieve high efficiency; (2) Develop propulsion systems with large numbers of thrusters and control systems; or (3) Advance the state of the art of ion thrusters to much larger size per thruster.

Nuclear power for electric propulsion - Space power reactor technology now under development (SP-100) may be adequate; needed advances are modest. Advanced power conversion systems are required to obtain power-to-mass ratios of interest. The SP-100 baseline is thermoelectric, which has no hope of meeting propulsion system performance needs. The most likely candidates are the closed Brayton (gas) cycle and the potassium Rankine (liquid/vapor) cycle. (Potassium provides the best match of liquid/vapor fluid properties to desired cycle temperatures.) Stirling cycle, thermionics, and a high- temeperature thermally-driven fuel cell are possibilities. The basic technology for Brayton and Rankine cycles are mature; both are in widespread industrial use. Prototype space power Brayton and Rankine turbines have run successfully for thousands of hours in laboratories. The development risk here is that these are very complex systems; there is no experience base for coupling a space power reactor to a dynamic power conversion cycle; there is no space power experience base at the power levels needed; and these systems, at power levels of interest for SEI space transfer application, are large enough to require in-space assembly and checkout. Space welding will be required for fluid systems assembly.

Solar power for space transfer propulsion - Solar power systems for space propulsion must attain much higher power-to-mass ratios than heretofore achieved. This implies a combination of advanced solar cells, probably multi-band-gap, and lightweight structural support systems. Required array areas are very large. Low-cost arrays, e.g. $100/watt, are necessary for affordable system costs, and automated construction of the large area structures, arrays, and power distribution systems appears also necessary. Where the nuclear electric systems are high development risk because of complexity and the lack of experience base at relevant power levels and with the space power conversion technologies, most of the solar power risk appears as technology advancement risk. If the technology advancements can be demonstrated, development risk appears moderate.

Avionics and software - Avionics and software requirements for space transfer systems are generally within the state of the art. New capability needs are mainly in the area of vehicle and subsystem health monitoring. This is in part an integration problem, but new technoques such as expert and neural systems are likely to play an important role.

An important factor in avionics and software development is that several vehicle elements having similar requirements will be developed, some concurrently. A major reduction in cost and integration risk for avoinics can be achieved by advanced development of a "standard" avionics and software suite, from which all vehicle elements would depart.

Further significant cost savings are expected from advancements in software development methods and environments.

Environmental Control and Life Support (ECLS) - The main development risk in ECLS is for the Mars transfer habitat system. Other SEI space transfer systems have short enough operating durations that shuttle and Space Station Freedom ECLS system derivatives will be adequate. The Mars transfer requirement is for a highly closed physico-chemical system capable of 3 years' safe and dependable operation without resupply from Earth. The development risk arises from the necessity to demonstrate long life operation with high confidence; this may be expensive in cost and development schedule.

Man-Rating Approach

Man-rating includes three elements: (1) Design of systems to manned flight failure tolerance standards, (2) Qualification of subsystems according to normal man-rating requirements, and (3)

Flight demonstration of critical performance capabilities and functions prior to placing crews at risk. Several briefing charts follow: the first summarizes a recommended approach and lists the subsystems and elements for which man-rating is needed; subsequent charts present recommended man-rating plans.

Mission and Operations Risk

These risk categories include Earth launch, space assembly and orbitasl launch, launch windows, mission risk, and mitigation of ionizing radiation and zero-g risks.

Earth launch - The Earth launch risk to in-space transportation is the risk of losing a payload because of a launch failure. Assembly sequences are arranged to minimize the impact of a loss, and schedules include allowances for one make-up launch each mission opportunity.

Assembly and Orbital Launch Operations - Four sub-areas are covered: assembly, test and on-orbit checkout, debris, and inadvertent re-entry.

Assembly operations risk is reduced by verifying interfaces on the ground prior to launch of elements. Assembly operations equipment such as robot arms and manipulators will undergo space testing at the node to qualify critical capabilities and performance prior to initiating assembly operations on an actual vehicle.

Assembly risk varies widely with space transfer technology. Nuclear thermal rocket vehicles appear to pose minimum assembly risk; cryo/aerobraking are intermediate, and nuclear and solar electric systems pose the highest risk.

Test and on-orbit checkout must deal with consequences of test failures and equipment failures. This risk is difficult to quantify with the present state of knowledge. Indications are: (1) large space transfer systems will experience several failures or anomalies per day. Dealing with failures and anomalies must be a routine, not exceptional, part of the operations or the operations will not be able to launch space transfer systems from orbit; (2) vehicles must have highly capable self-test systems and must be designed for repair, remove and replace by robotics where possible and for ease of repair by people where robotics cannot do the job; (3) test and on-orbit checkout will run concurrently with propellant loading and launch countdowns. These cannot take place on Space Station Freedom. Since the most difficult part of the assembly, test and checkout job must take place off Space Station Freedom the rest of the job probably should also.

Orbital debris presents risk to on-orbit operations. Probabilities of collision are large for SEI-class space transfer systems in low Earth orbit for typical durations of a year or more. Shielding is mandatory. The shielding should be designed to be removed before orbital launch and used again on the next assembly project.

Creation of debris must also be dealt with. This means that (1) debris shielding should be designed to minimize creation of additional debris, especially particles of dangerous size, and (2) operations need to be rigorously controlled to preventinadvertent loss of tools and equipment that will become a debris hazard.

Inadvertent reentry is a low but possible risk. Some of the systems, especially electric propulsion systems, can have very low ballistic coefficient and therefore rapid orbital decay rate. Any of the SEI space transfer systems will have moderately low ballistic coefficient when not loaded with propellant. While design details are not far enough along to make a quantitative assessment, parts of these vehicles would probably survive reentry to become ground impact hazards in case of inadvertent reentry. For nuclear systems, it will be necessary to provide special support systems and infrastructure to drive the probability of inadvertent reentry to extremely low levels.

Launch Windows - Launch windows for single-burn high-thrust departures from low Earth orbit are no more than a few days because regression of the parking orbit line of nodes causes relatively rapid misalignment of the orbit plane and departure vector. For lunar missions, windows recur at about 9-day intervals.

For Mars, the recurrence is less frequent, and the interplanetary window only lasts 30 to 60 days. It is important to enable Mars launch from orbit during the entire interplanetary window. Three-impulse Mars departures make this possible; a plane change at apogee of the intermediate parking orbit provides alignment with the departure vector. Further analysis of the three-burn scheme is needed to assess penalties and identify circumstances where it does not work.

Launch window problems are generally minimal for low-thrust (electric propulsion) systems.

Mission Risk - Comparative mission risk was analyzed by building risk trees and performing semi-quantitative analysis. The next chart presents a comparison of several mission modes; after that are the risk trees for these modes.

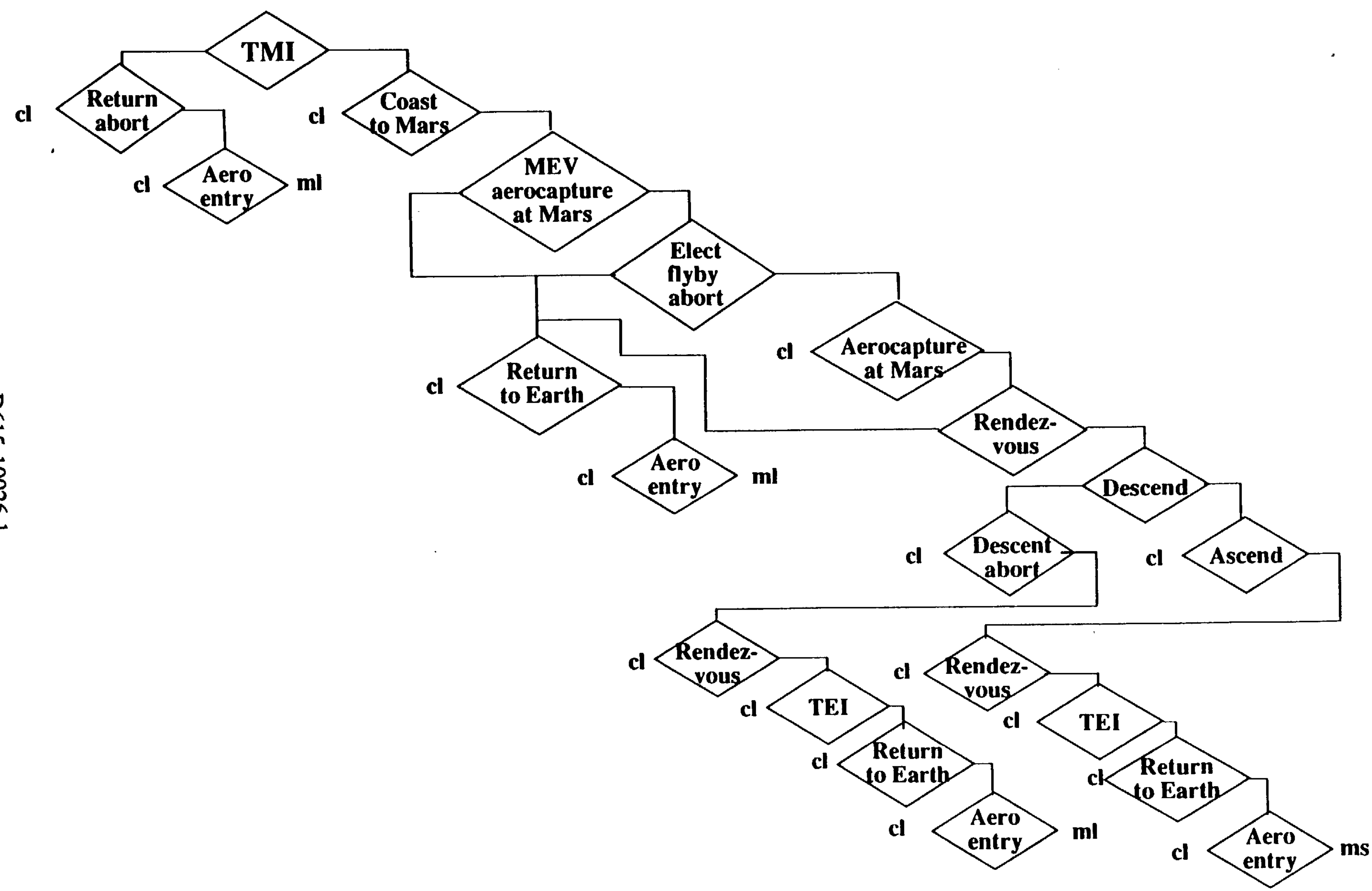
Ionizing Radiations and Zero G - The threat from ionizing radiations is presented elsewhere in this document. Presented here are the mitigating strategies for ionizing radiations and zero g.

Nuclear systems operations present little risk to flight crews. Studies by University of Texas at Austin showed that radiation dose to a space station crew from departing nuclear vehicles is very small provided that sensible launch and flight strategies are used. On-board crews are protected by suitable shieding and by arrangement of the vehicle, i.e. hardware and propellant between reactors and the crew and adequate separation distances. After nuclear engines are shut off, radiation levels drop rapidly so that maneuvers such as departure or return of a Mars excursion vehicle are not a problem. On-orbit operations around a returned nuclear vehicle are deferred until a month or two after shutdown, by which time radioactivity of the engine is greatly reduced.

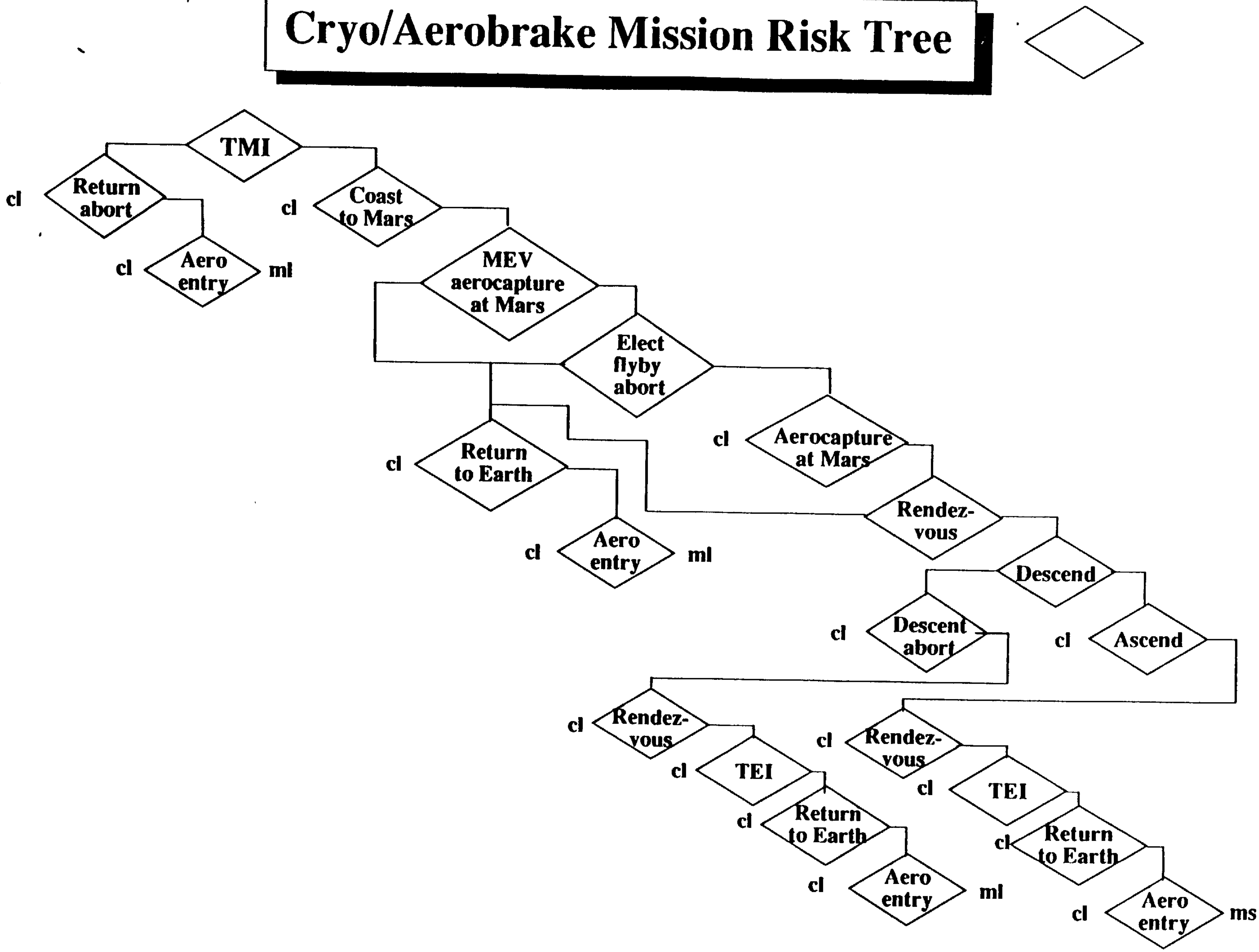
Reactor disposal has not been completely studied. Options include solar system escape and parking in stable heliocentric orbits between Earth and Venus.

Crew radiation dose abatement employs "storm shelters" for solar flares, and either added shielding of the entire vehicle or fast transfers (or both) to reduce galactic cosmic ray exposure. Assessments are in progress; tradeoffs of shielding versus fast trips have yet to be completed. Expected impact for lunar missions is negligible and for Mars missions, modest.

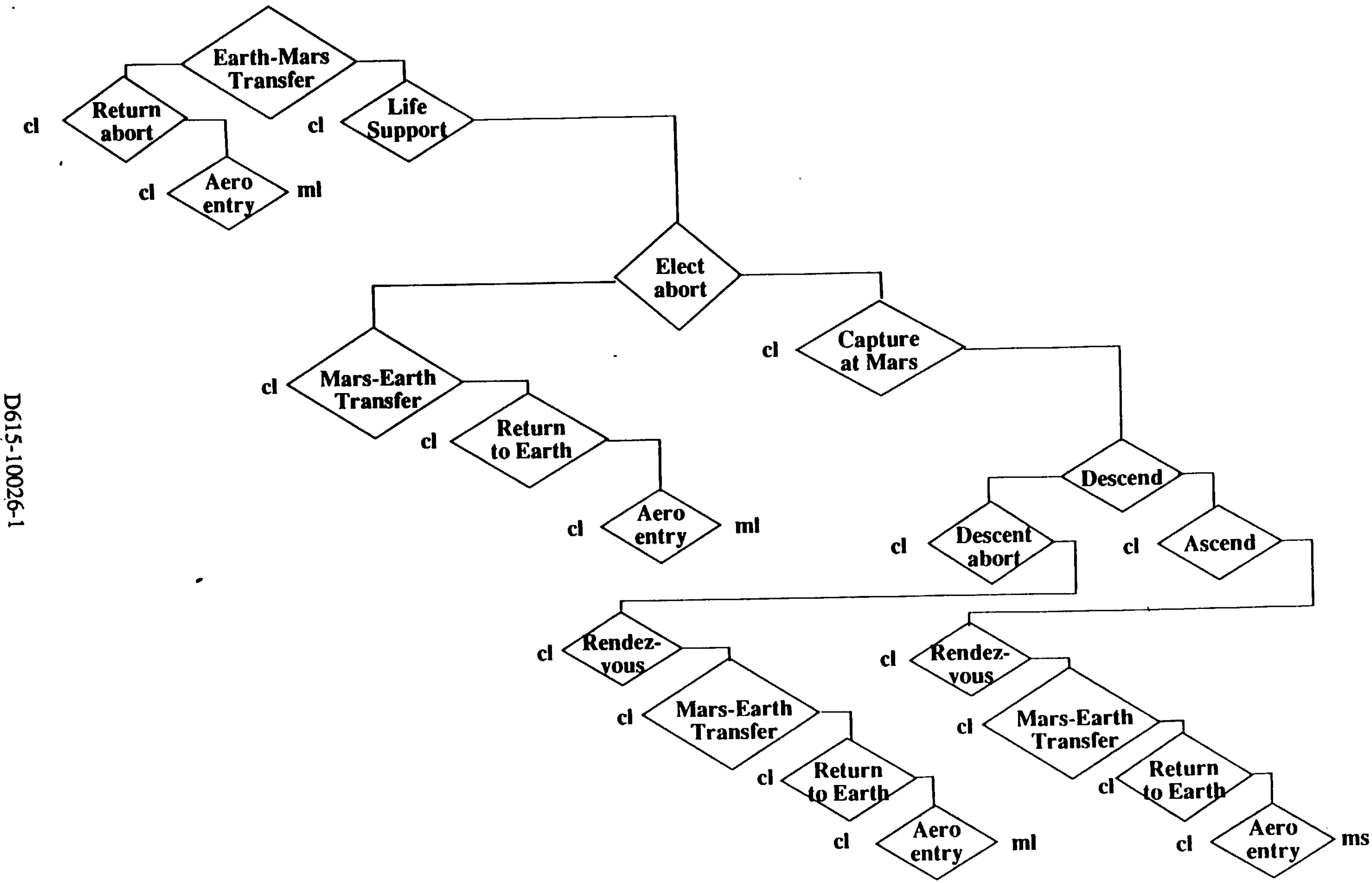
Cryo/Aerobrake Mission Risk Tree



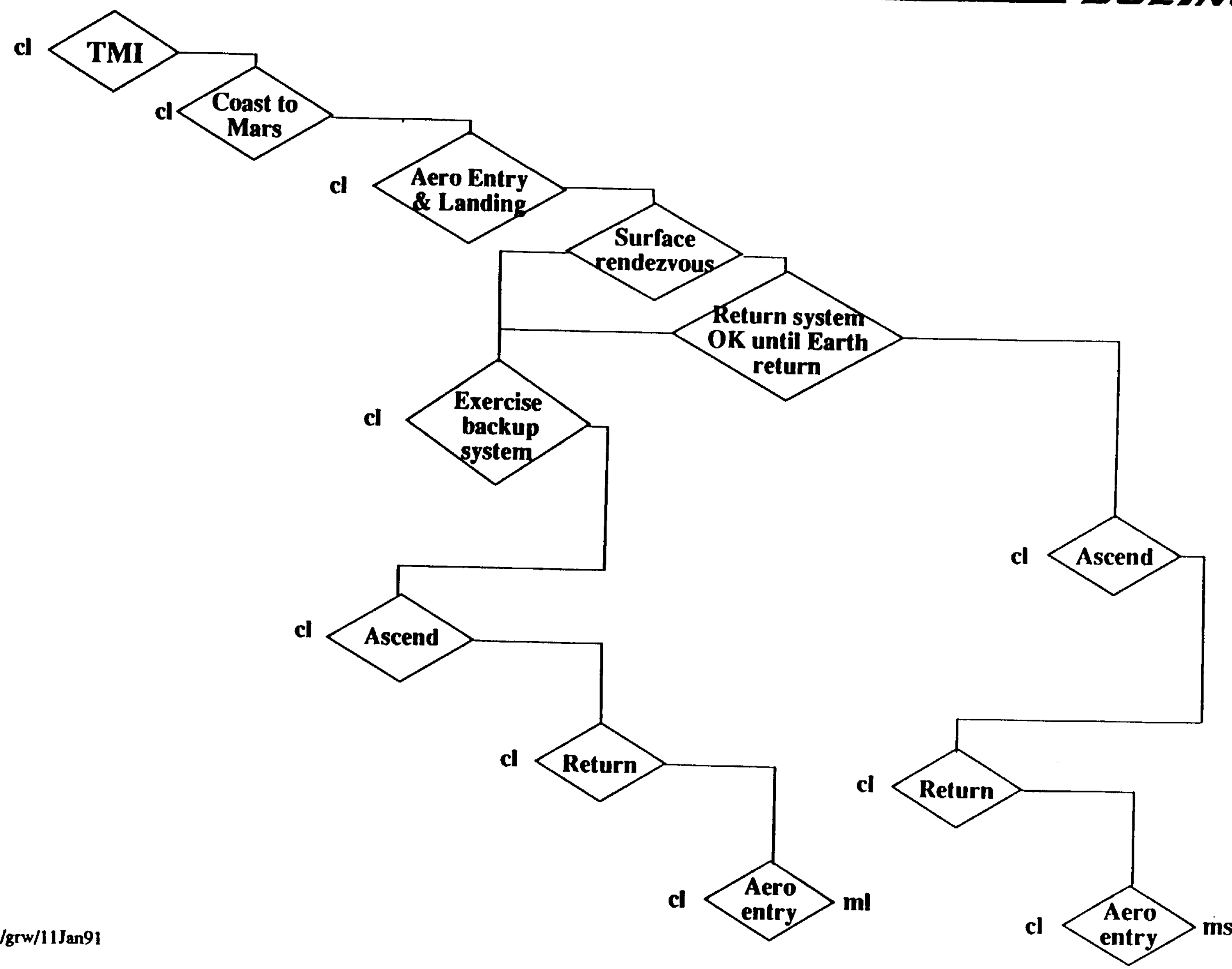
Cryo/Aerobrake Mission Risk Tree



Nuclear Electric Mission Risk Tree



Mars Direct Mission Risk Tree - Crew Mission



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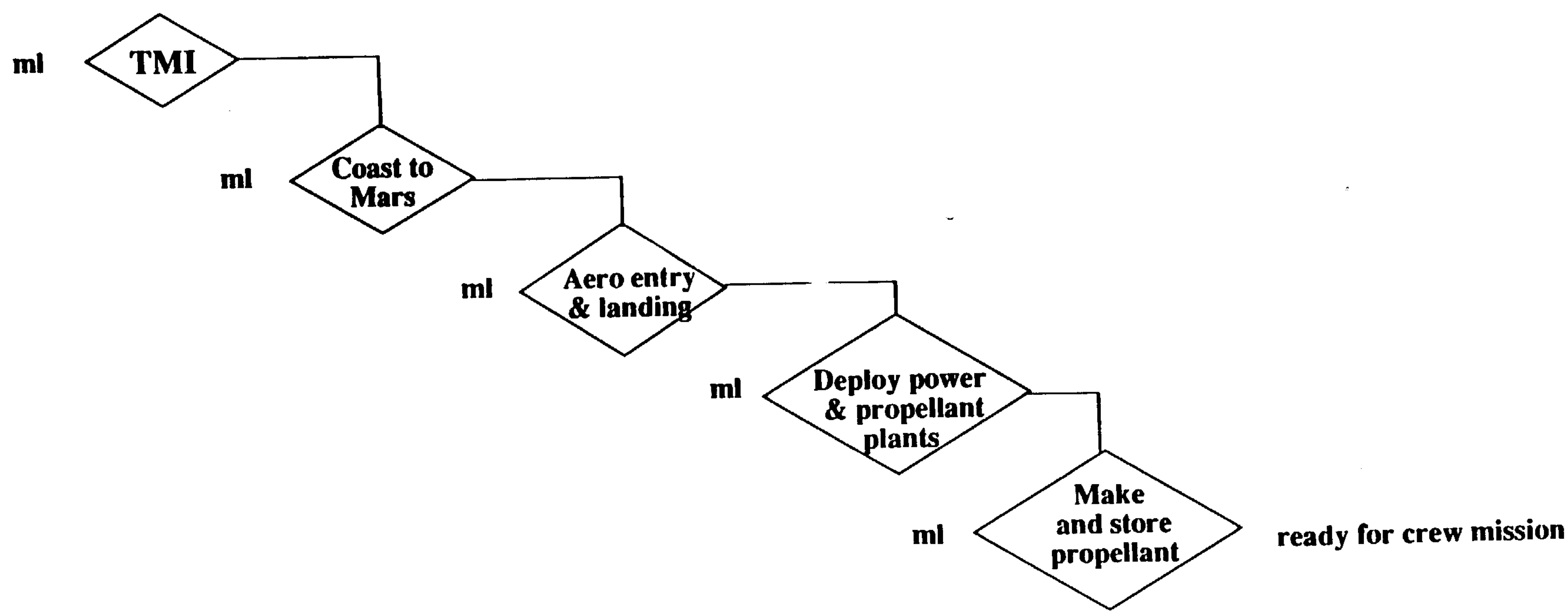
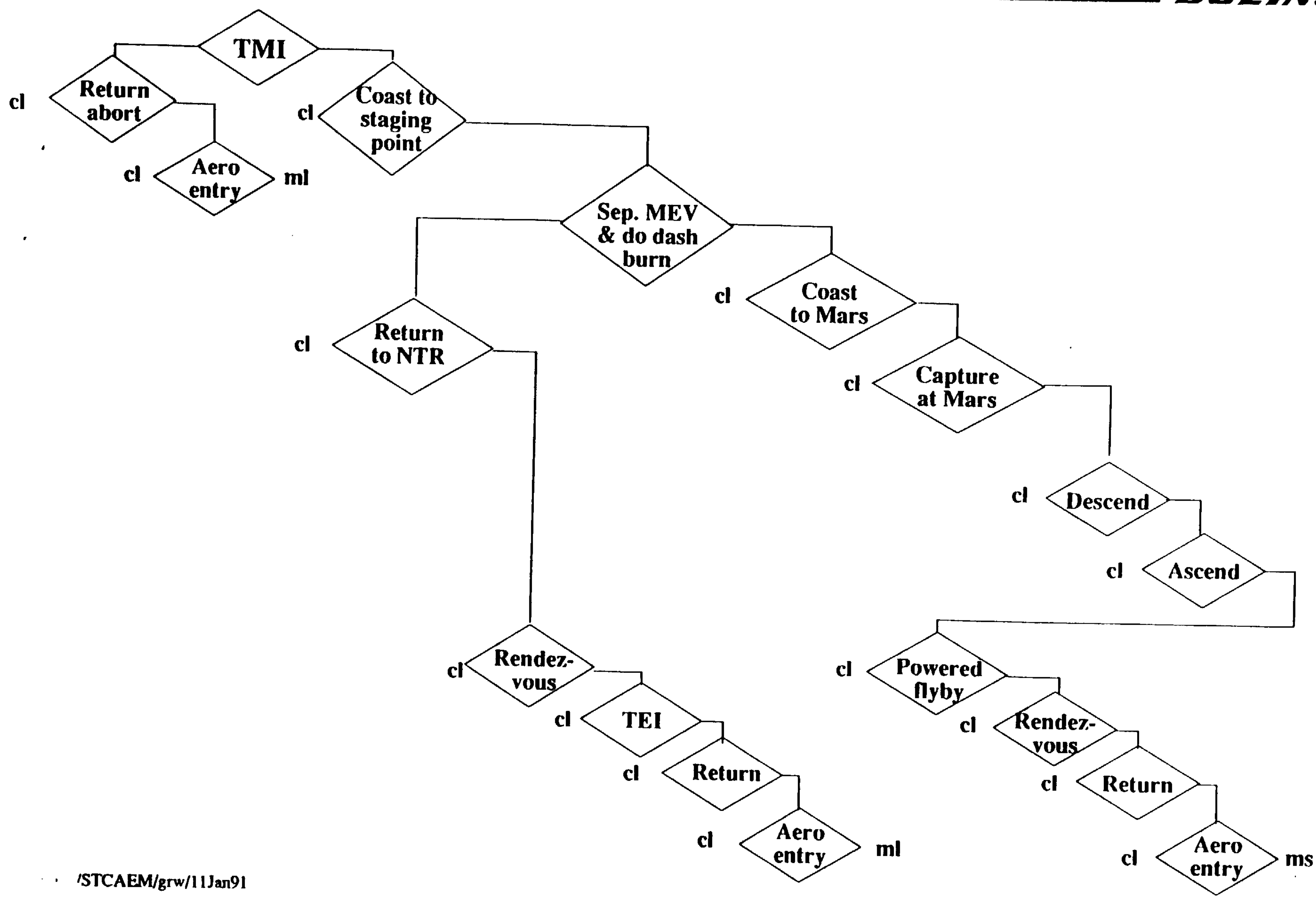
Mars Direct Mission Risk Tree - Cargo Mission



Nuclear Rocket Dash Mode Mission Risk Tree



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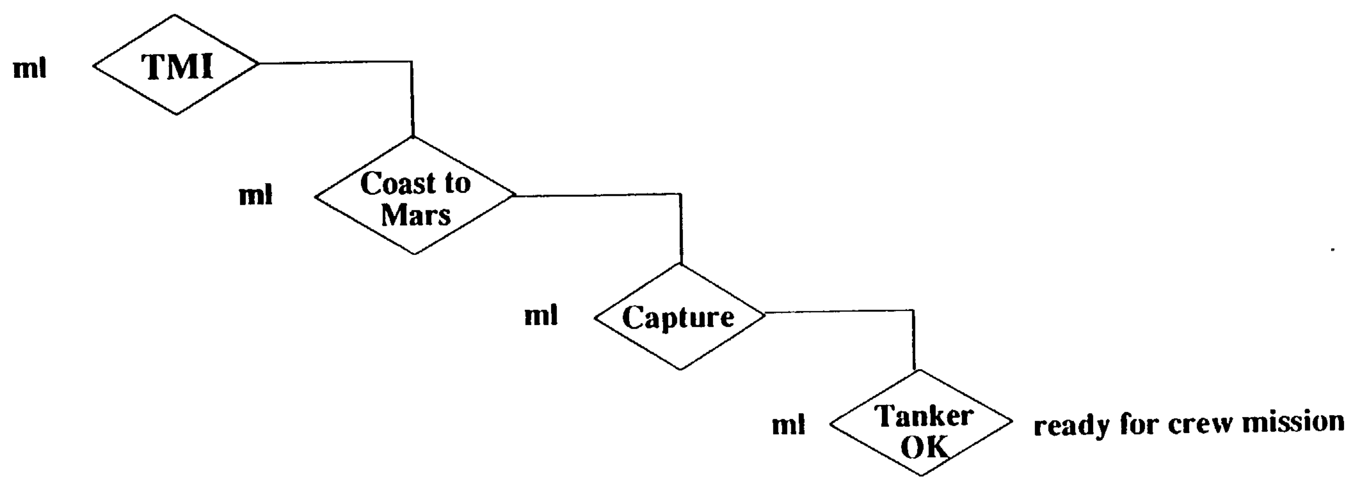
Nuclear Rocket Split Sprint Mission Risk Tree
Cargo Mission



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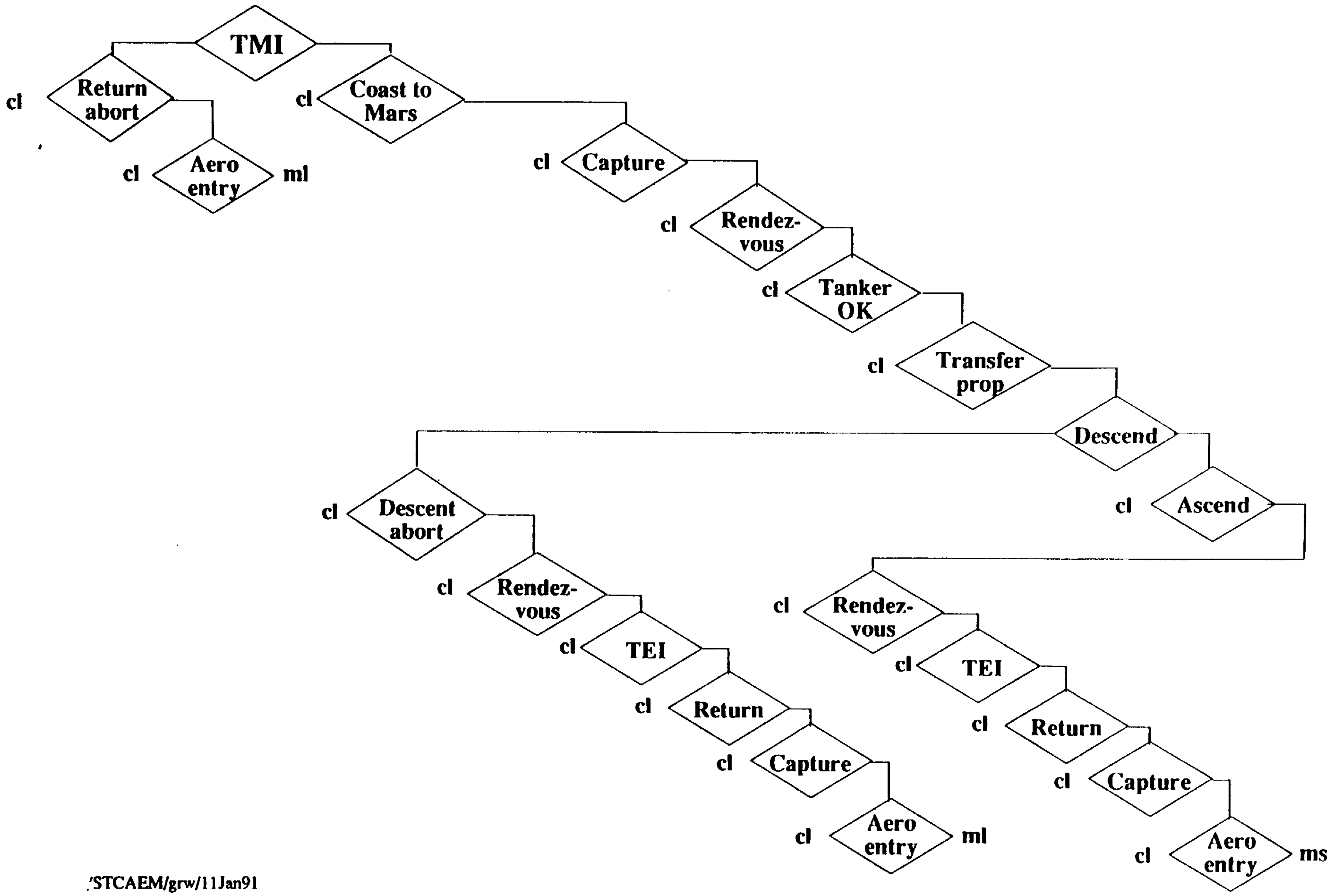
Nuclear Rocket Split Sprint Mission Risk Tree
Crew Mission



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Man Rating Requirements

The facing page describes our recommended approach to man-rating and lists the systems/ subsystems for which we believe man-rating is required. Following pages present recommendations for man-rating programs for all but four of the systems. Three of those not presented (Crew modules/hab systems, Vehicle power, and Surface transportation) are judged suited for man-rating by normal space qualification means. If advanced technologies are adopted for any of these, a specific overall man-rationg program should be defined.

The fourth is not presented because there are basic questions as to overall approach. This is the need for vehicle health monitoring and on-board maintenance systems. This is a mix of subsystems technology and health monitoring and diagnostics technology. A key issue here is to select the general technical approach: (a) what kinds of sensors, (b) what kidns of subsystem models, (c) what kinds of logic (fault detection by sensors vs. inference of failures through system/subsystem state comparison with math models of normal and degraded performance), and (d) what new technologies (expert systems; neural nets) and how these can be integrated into an overall health maintenance architecture and validataed for safe and successful use on long-duration missions. The need goes beyond health monitoring and diagnostics; it is also necessary to develop an on-board maintenance system that can instruct the crew how to perform maintenance and how to perform any testing not built in to the health monitoring system. Demonstration of the overall capability for Mars missions is included on the Mars avionics chart.

Man-Rating Approaches

A set of charts present recommended man-rating approaches for aerobrakes, cryogenic rocket engines, nuclear rocket engines, cryogenic propellant system, auxiliary (attitude control) propulsion systems, nuclear and solar electric propulsion systems, ECLSS, and Avionics and communications systems.

The aerobraking approach makes use of the lunar tandem LTV booster as a full-size lunar aerobrake testbed, together with reliance on ground-test facilities and CFD. The Mars aerobrake is qualified on an unmanned cargo delivery mission.

The cryogenic rocket engine program is a conventional one of technology demonstration, flight hardware qualification, and flight demonstration.

A sequence of major tests and demonstrations to achieve nuclear rocket man-rating is shown next. Note that two flight demonstration options exist. A decision of which to use depends on whether cargo delivery to Mars is needed before the first manned mission, as would be the case if a conjunction fast transfer and long surface stay is required on the first mission to reduce galactic cosmic ray exposure to the crew.

The cryogenic propellant sequence relies on the fact that the STCAEM baseline initial lunar system, tandem-direct, does not need zero-g propellant transfer or gauging; this allows more time for development of these challenging technologies.

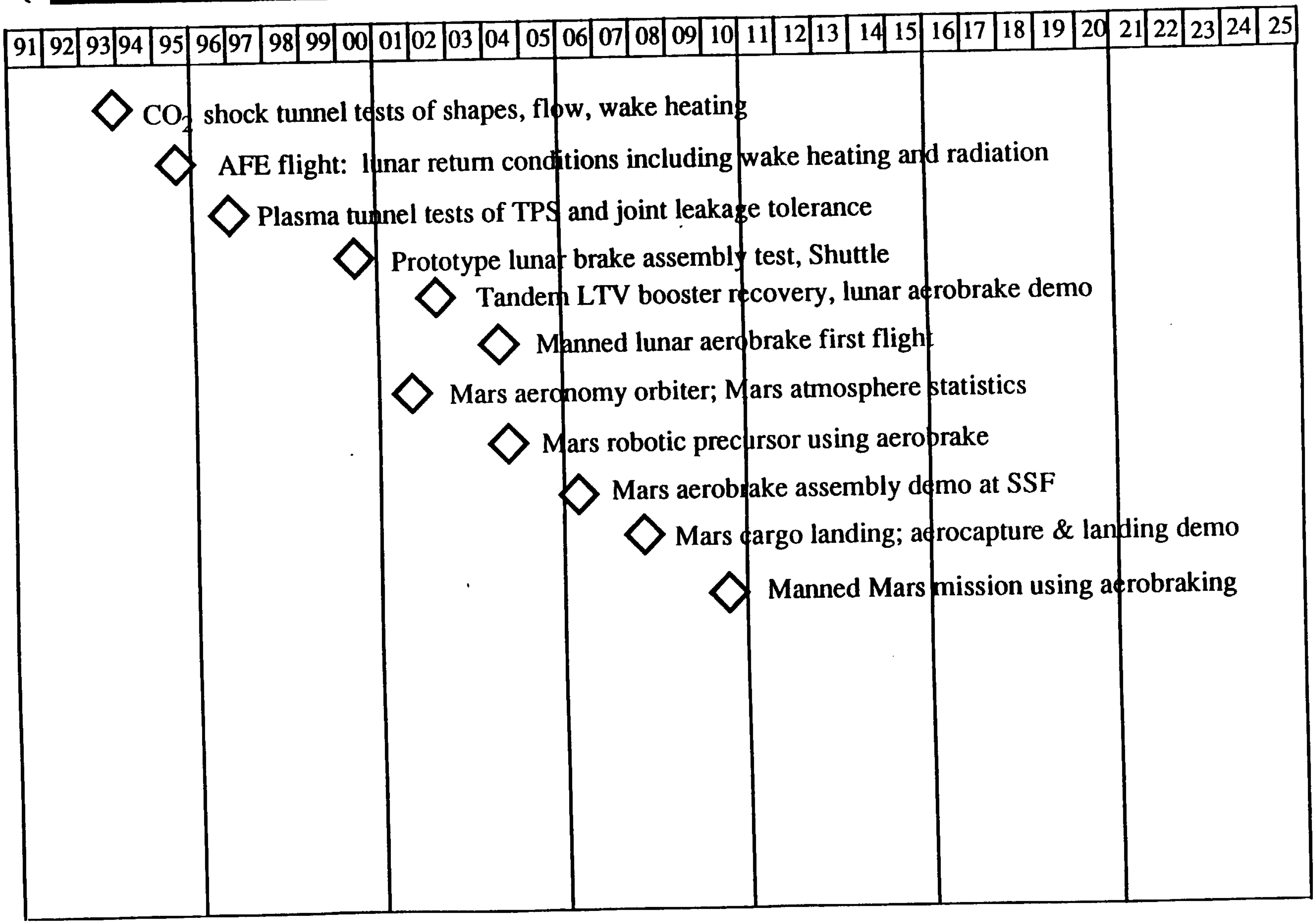
The auxiliary propulsion assumes that an advanced technology using cryogenic propellants from main tanks is adopted. If conventional storable propulsion technology is used, a conventional qualification program is sufficient. Storable technology is mature enough that flight demonstration is not needed.

The nuclear and solar electric program is the most complex depicted here, in part because in includes both nuclear and solar power generation technology. As presented here, a choice between nuclear and solar power generation is made in the late 1990s after technology demonstrations.

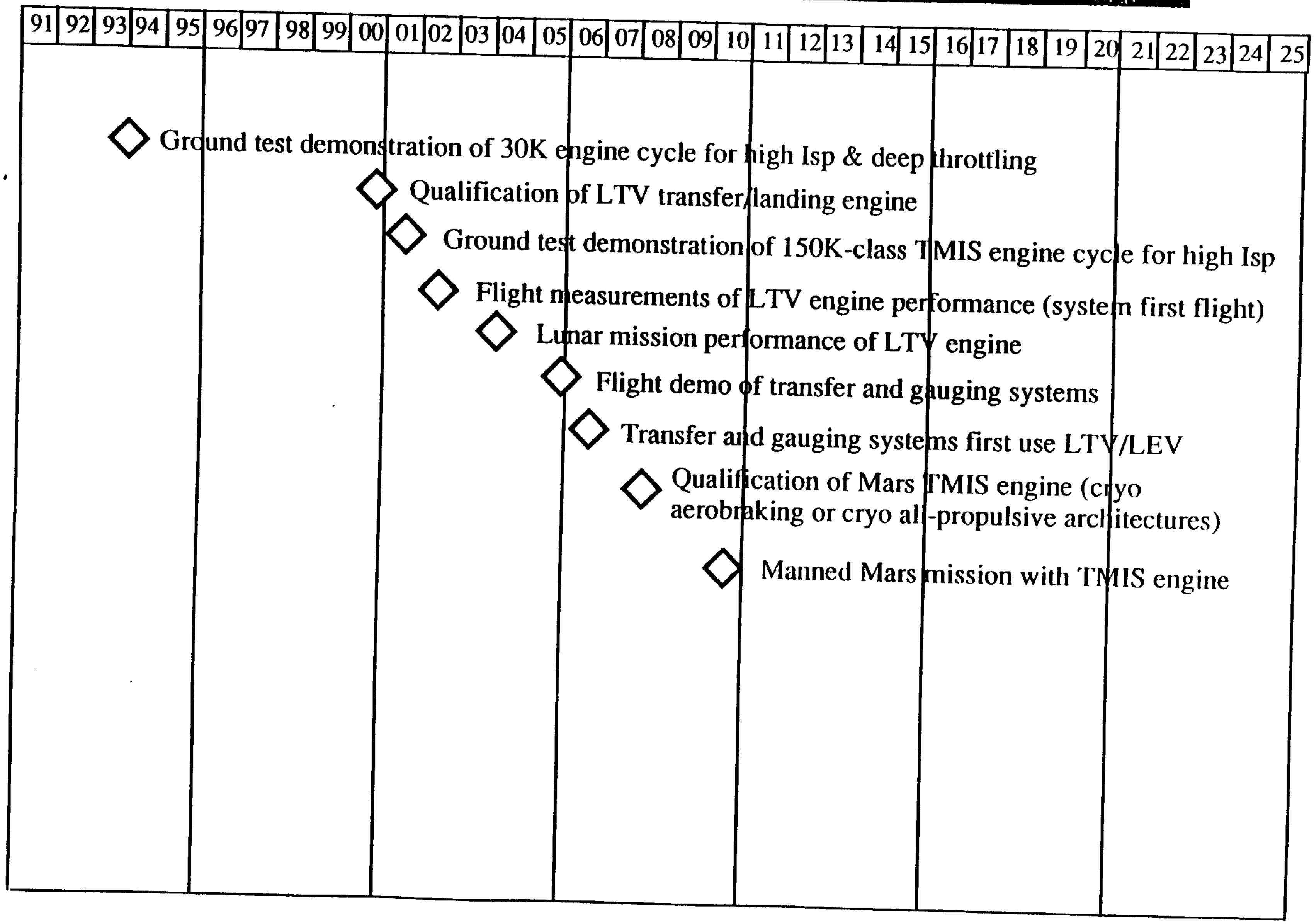
ECLSS man-rating includes conventional physico-chemical and CELSS systems. Whether CELSS benefits long-duration transportation system crew habitats has yet to be determined. This presentation assumes that an integrated ECLSS development for surface and transportation systems will occur.

The avionics program presumes a standard avionics architecture with unique appendages for unique requirements such as Mars aerocapture GN&C.

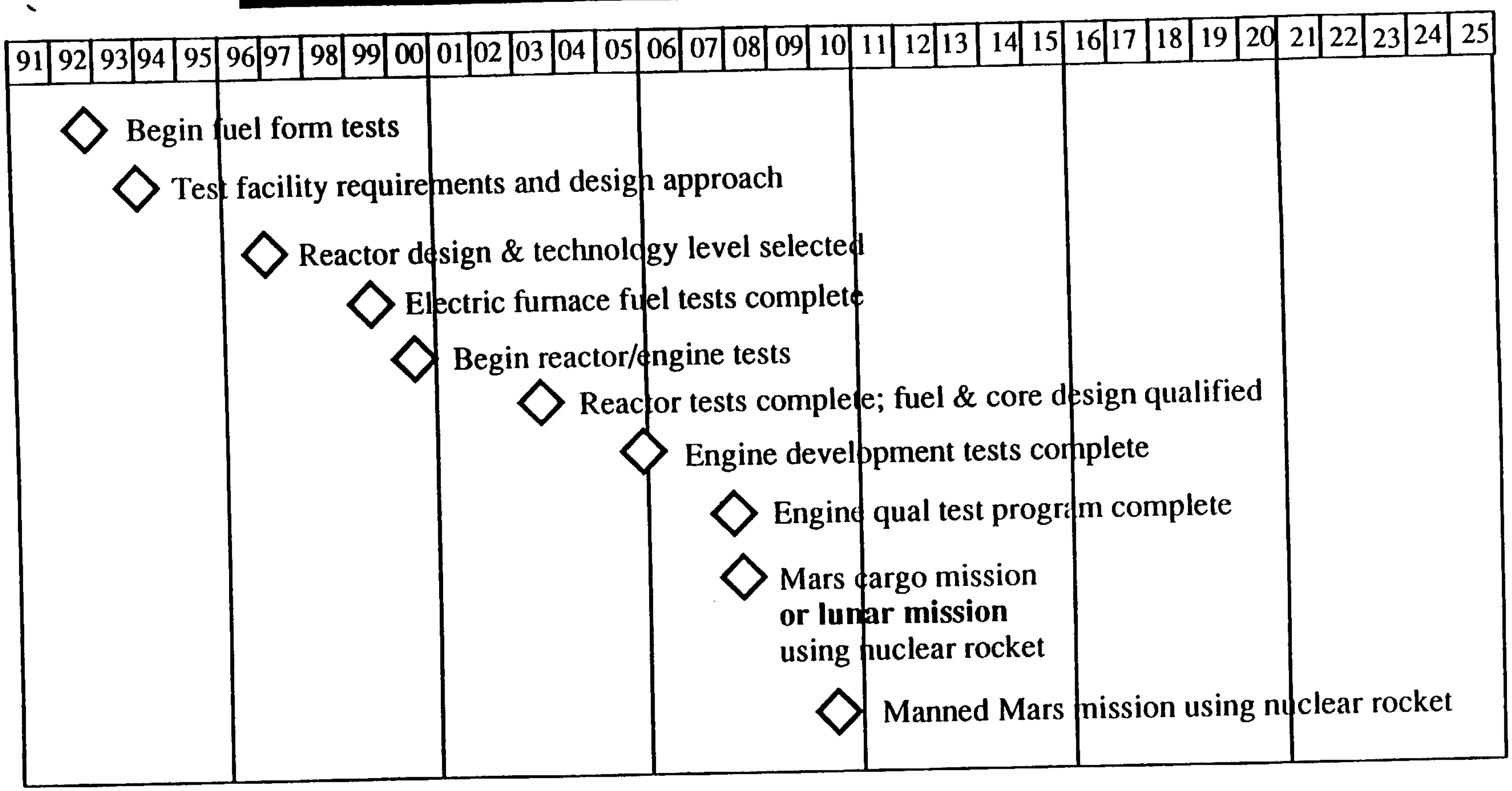
Aerobraking Major Test/Demo Man-Rating Approach



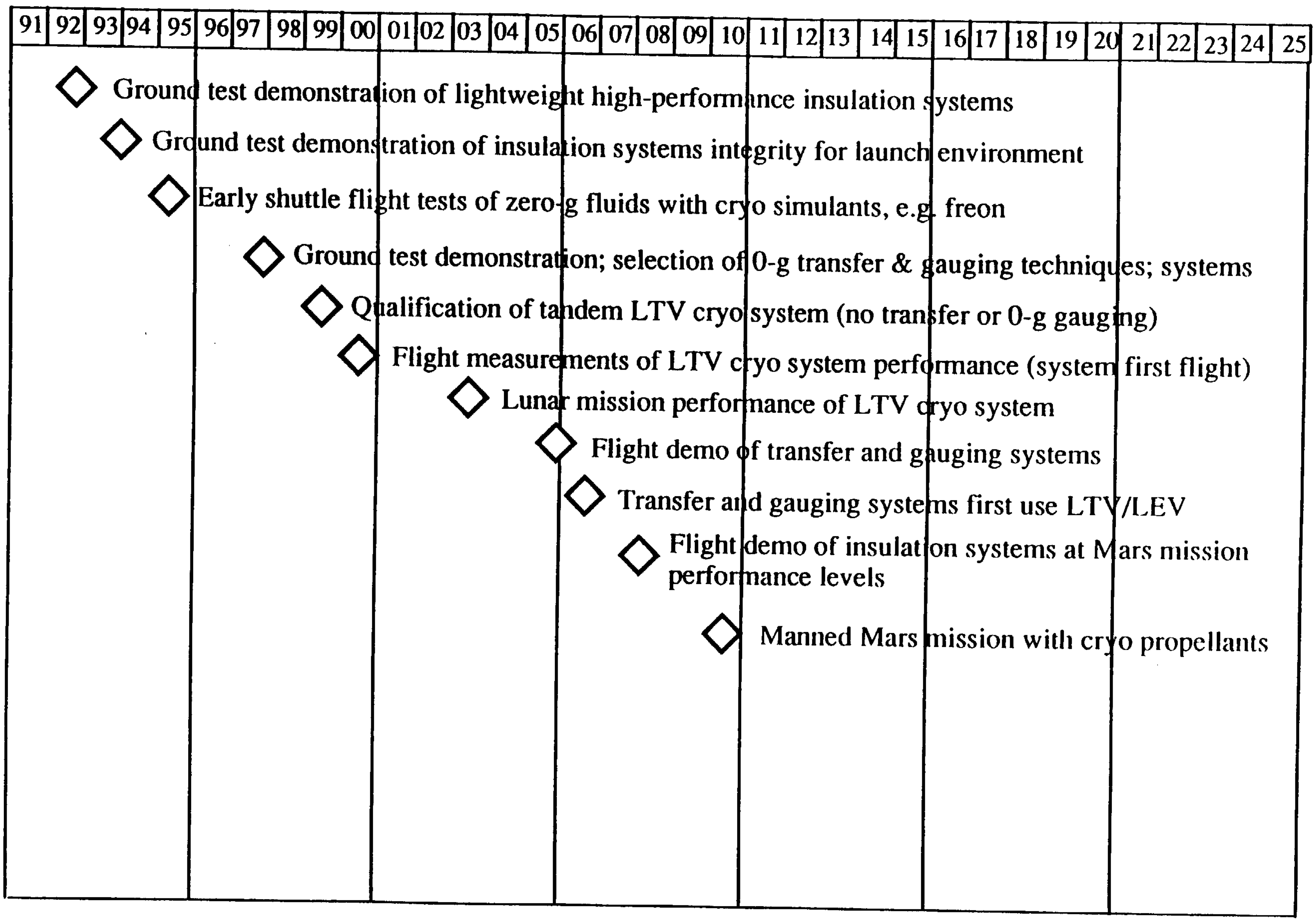
Cryogenic Rocket Engine Man-Rating Approach



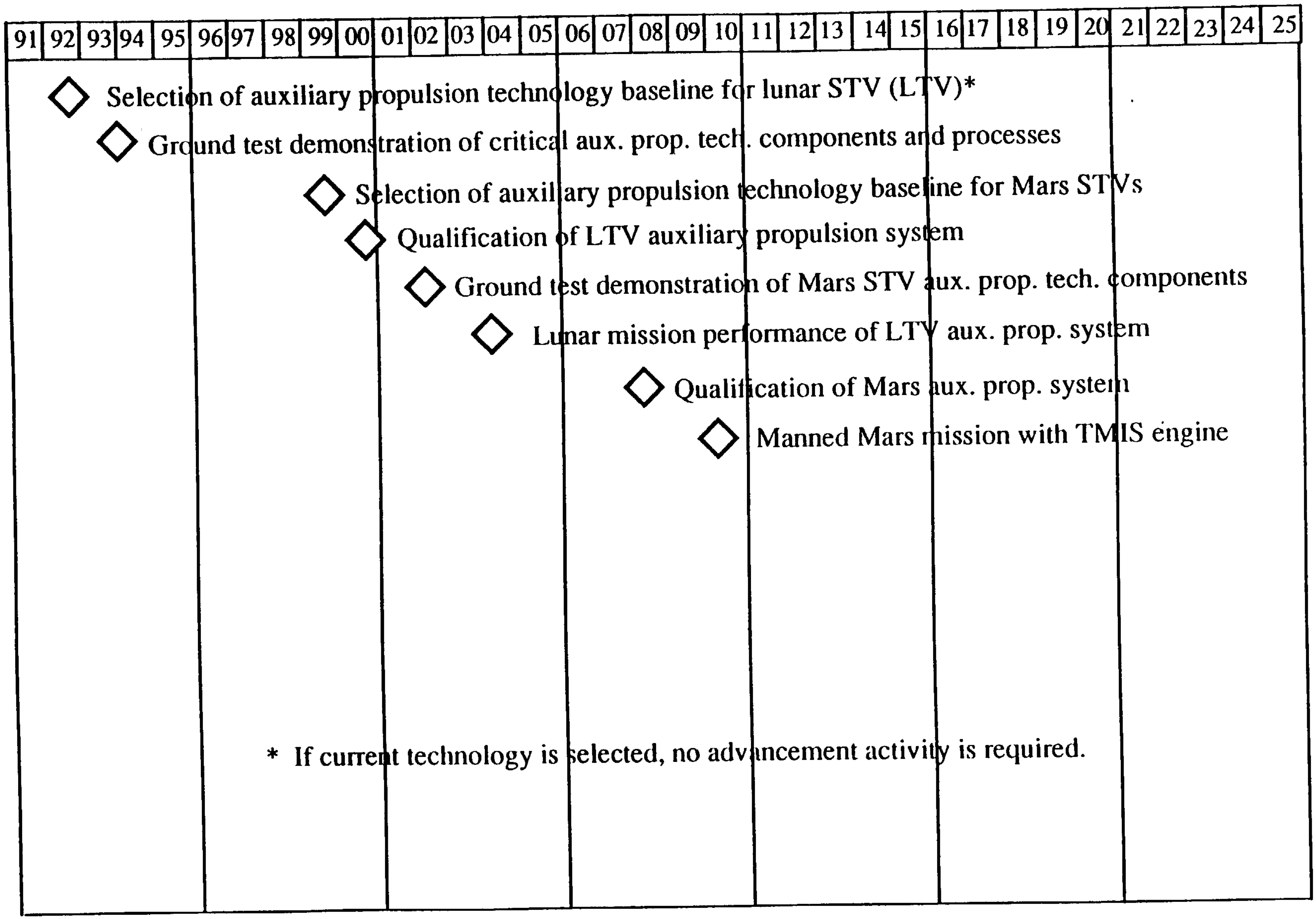
Nuclear Rocket Man-Rating Approach



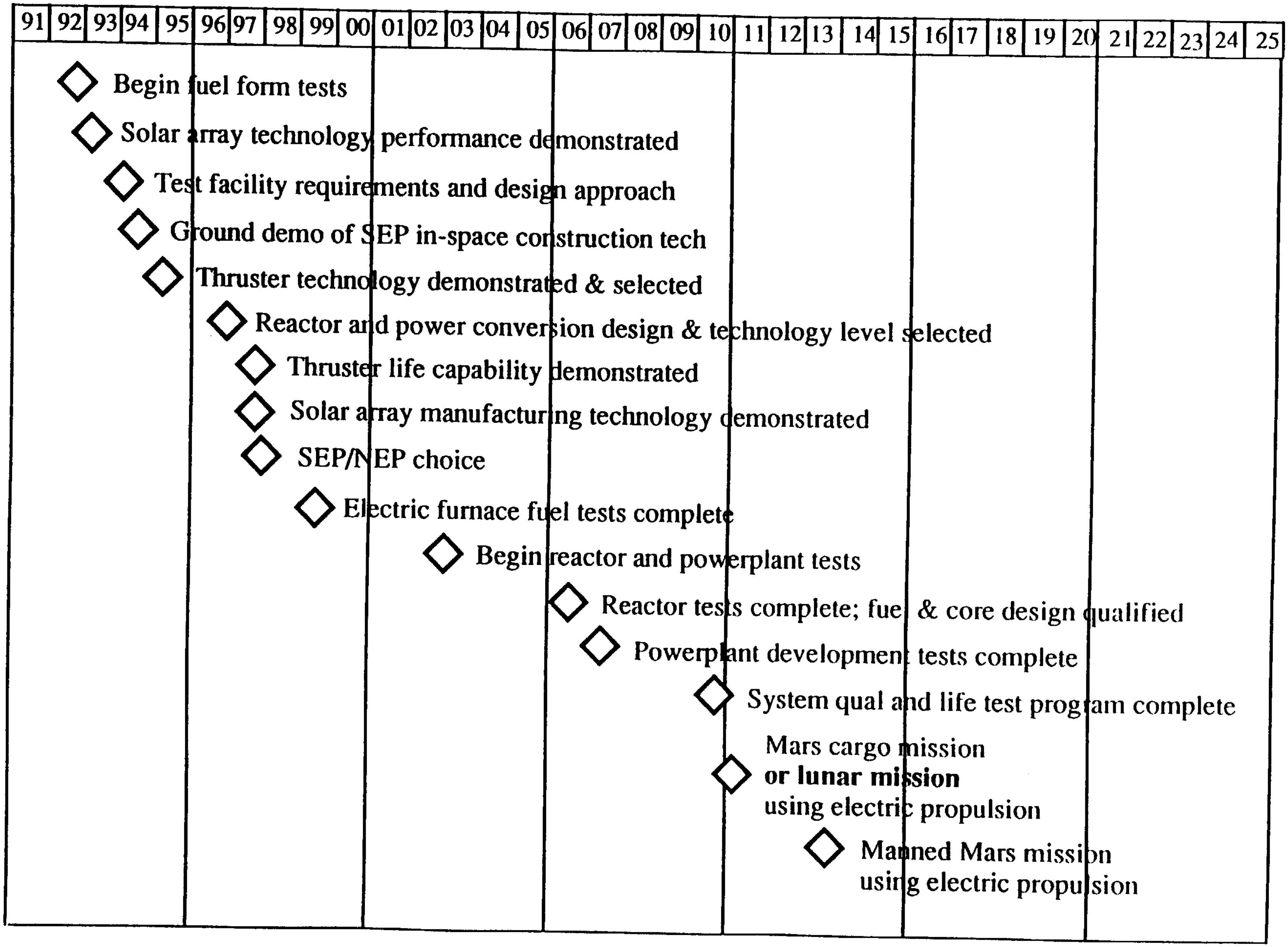
Cryogenic Propellant System Man-Rating Approach



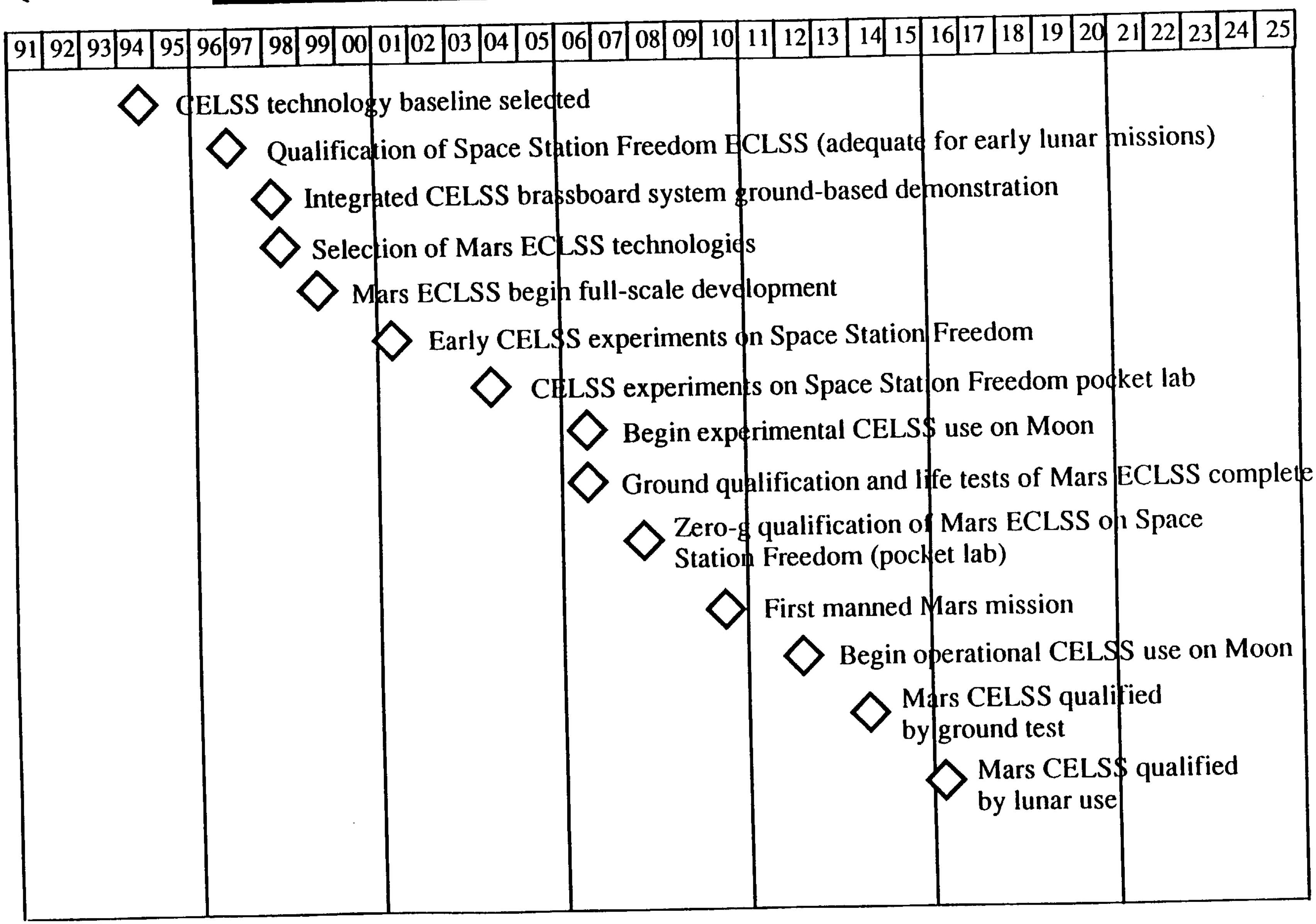
Advanced Auxiliary Propulsion Man-Rating Approach



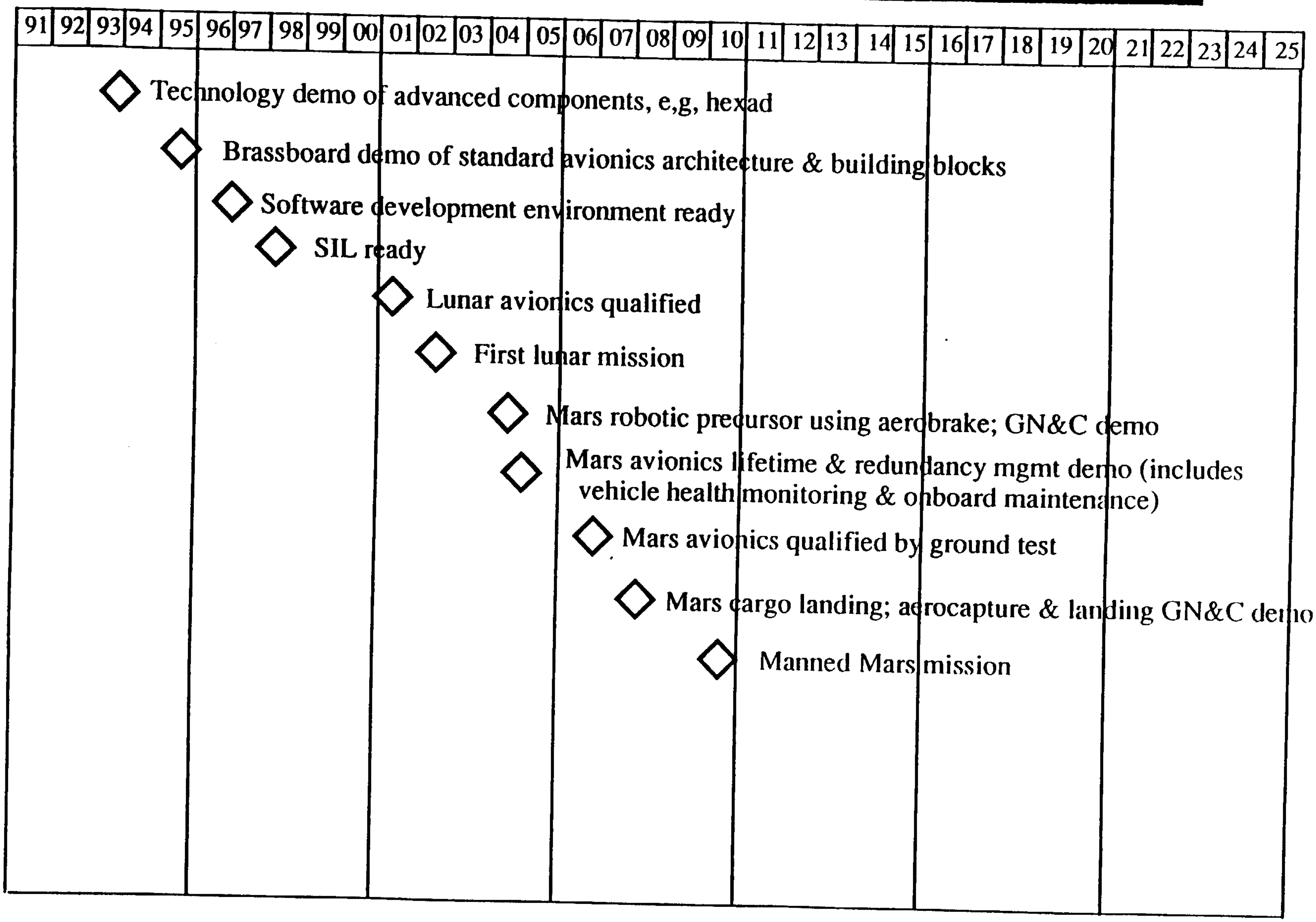
Nuclear and Solar Electric Propulsion Man-Rating Approach



ECLSS Systems Man-Rating Approach



Avionics Major Test/Demo Man-Rating Approach



Technology Development Concerns and Schedules - Cryogenic / Aerobraked Vehicle

Critical technology development issues relating to the reference CAB vehicle are presented in this section. Where applicable, the same charts are also included in the NTR, NEP, and SEP IP&ED documents. The focus of this section will be to bring out the most important issues relating to the reference cryogenic vehicle, and to present preliminary technology development schedules for these issues. The issues are presented here in outline form, beginning with the most important, with accompanying schedules wherever possible.

Aerobraking (low & high energy)

The technology category which offers the most potential vehicle benefits but which presents the highest degree of technology development uncertainties, is the area of high and low energy aerobraking. This area presents a variety of issues for technology development including high strength to mass ratio structural materials, high temperature thermal protection systems, avionics, assembly and operations, hypersonic test facilities and computer codes, and Mars atmosphere prediction. High strength structural material options include metal matrix composite, organic matrix composite, and advanced carbon-carbon elements. Other structural considerations include load distribution and attachment of payload for aerocapture, and ETO launch and assembly of large structures. Thermal protection systems issues include low mass ablative and reradiative materials, and structure/TPS integration issues. The aerobrake maneuver will place considerable demands on the vehicle avionics system with the need for real time trajectory analysis, and vehicle guidance and control. The launch and assembly of the large aerobrake structure will present ground and space operations problems which will require technology and advanced development in both the areas of design and operations. Finally, computational analysis and atmosphere prediction capability will be critical in the development of a man-rated aerobrake for Mars use. A preliminary development schedule for Lunar and Mars aerobrake technology development is presented. It includes the major milestones for both ground and flight testing. The points where a Lunar and Mars full scale development decision can be made are also highlighted on the schedule.

Cryogenic Propulsion and Fluid Management

Cryogenic propulsion and long term fluid management technologies offer mission vehicle benefits over lower performance storable propulsion systems, comparable to those provided by high energy aerobraking. The high Isp of a LH_2-LOX system (460-480 s) can reduce vehicle IMLEO greater than 50% over the lower Isp (280-360 s with metallic gels) storable systems. The long term storage and low-g fluid management of cryogenic fluids, along with long lifetime, in-space restartable cryogenic engines are the major technology development concerns for a cryogenically fueled vehicle. Preliminary technology schedules are presented for space based cryogenic engines, and cryogenic fluid system development for both Lunar and Mars applications. The cryogenic space based engine development effort begins with the planned AETB work at LeRC, and continues on to development work for a large engine for Mars applications. The cryogenic fluid systems schedule includes Earth-based thermal control and selected management (tank pressure control, liquid acquisition device effectiveness, etc.) tests, as well as planned flight experiments to carry out system and subsystem validation tests.

Vehicle Avionics and Software

Although the technology readiness level of vehicle avionics and software is ahead of many of the other technology areas listed in some respects, the demands on the system in the areas of processing rate, accuracy, autonomous operation, and status/health monitoring will drive technology and advanced development in areas not fully defined at this point. Software requirements cannot be fully determined until the vehicle design is at a more finished stage than the current levels. A preliminary schedule for autonomous systems development is presented. The decision points for full scale development The communications system options can be more fully defined before a final vehicle design is produced, however. A technology development schedule for advanced communications is presented.

Life Support

A reliable, redundant long term life support system will be enabling for future exploration missions. The degree of closure of, and the reliability of the system are the major technology development concerns. Low-g human factors determination will also be an important technology consideration which will drive vehicle design. An integrated schedule of the major areas of the life support technology development task are presented. It includes radiation shielding and materials, regenerative life support, and EVA systems development. As before, the points where Lunar and Mars full scale development decisions can logically be made in the technology program are highlighted.

In-Space Assembly and Processing

The in-space assembly and processing of large space transfer vehicles will present a variety of technology advanced development challenges, particularly for the large LTV and MTV and MEV aerobrakes. As shown on the accompanying schedule, extensive ground tests must occur before any orbital work can be initiated. The vehicle designs will be driven to a large degree by the assembly facilities and technologies seen as being available during the vehicle buildup sequence.

Summary

As noted before, many of the identified critical and high leverage technology development issues are common across all four major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H_2, and possibly O_2 for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique cryo/AB technology issues include high energy aerobraking, and large advanced space engine advanced development. Enhancing technologies include cryogenic refrigeration (lander tanks), O_2-H_2 RCS, advanced in-space assembly techniques, higher Isp cryogenic engines, and advanced structural materials development.

ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules

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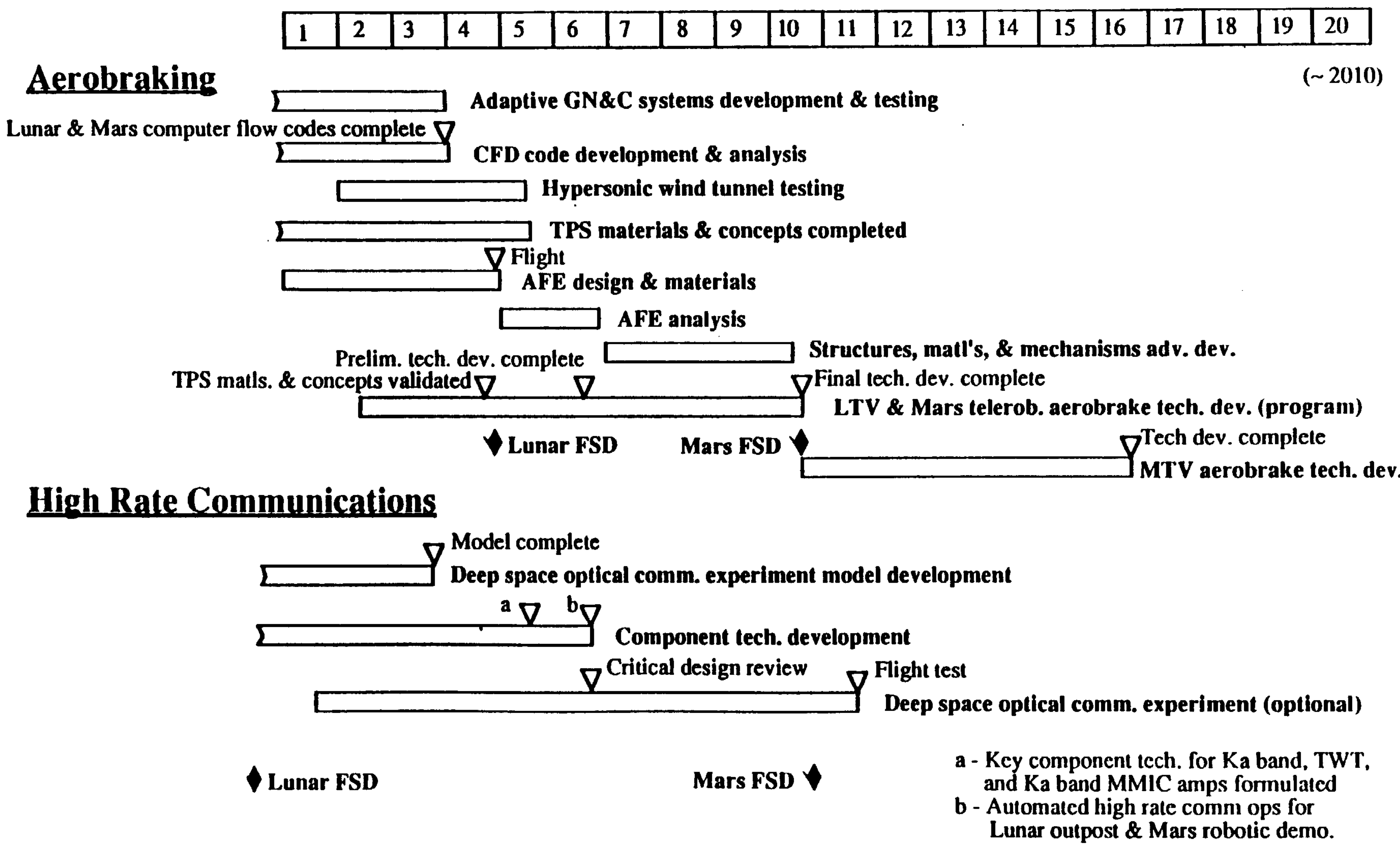

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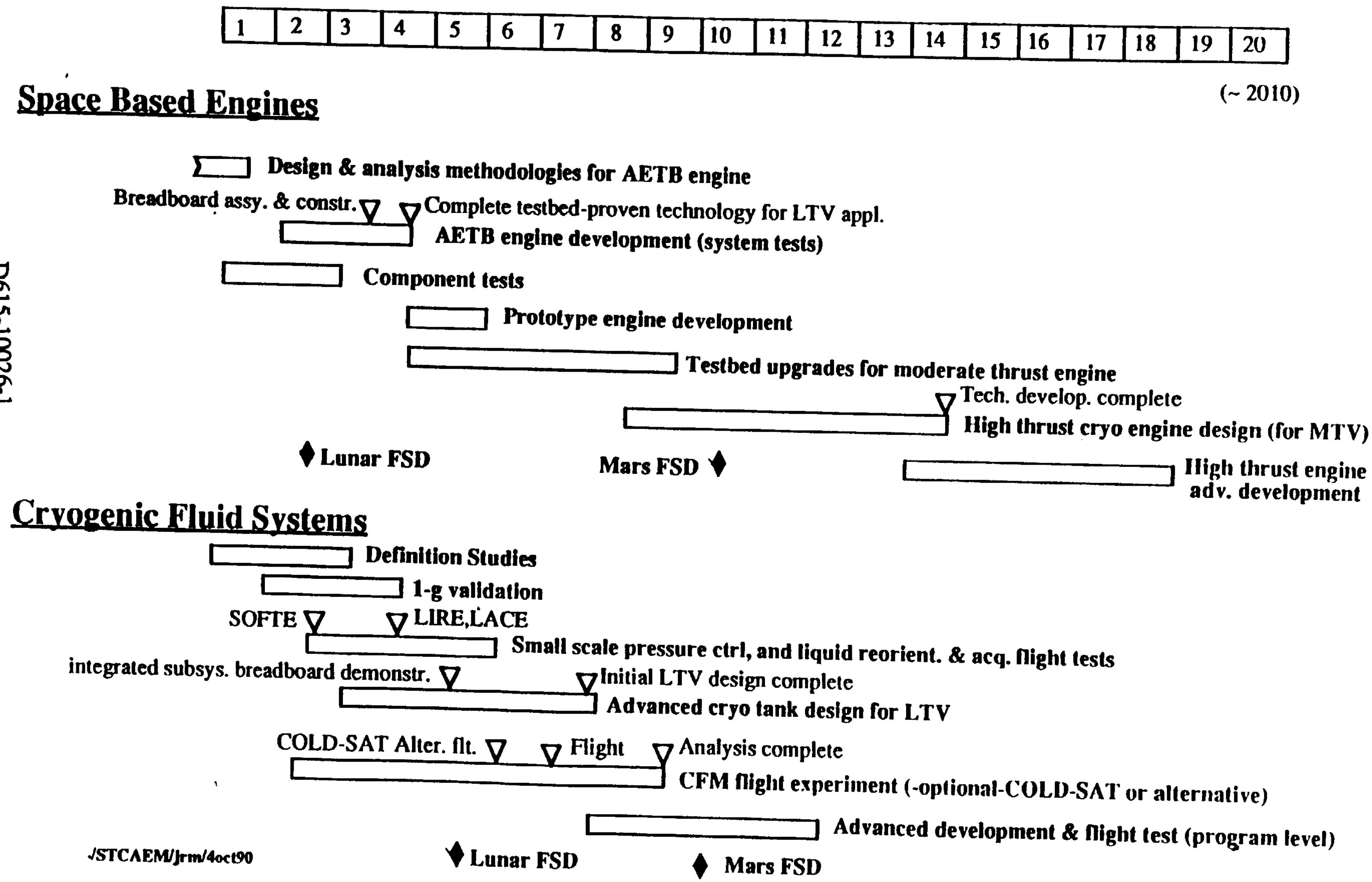
Preliminary SEI Technology Development Schedules(Cont.)



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ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

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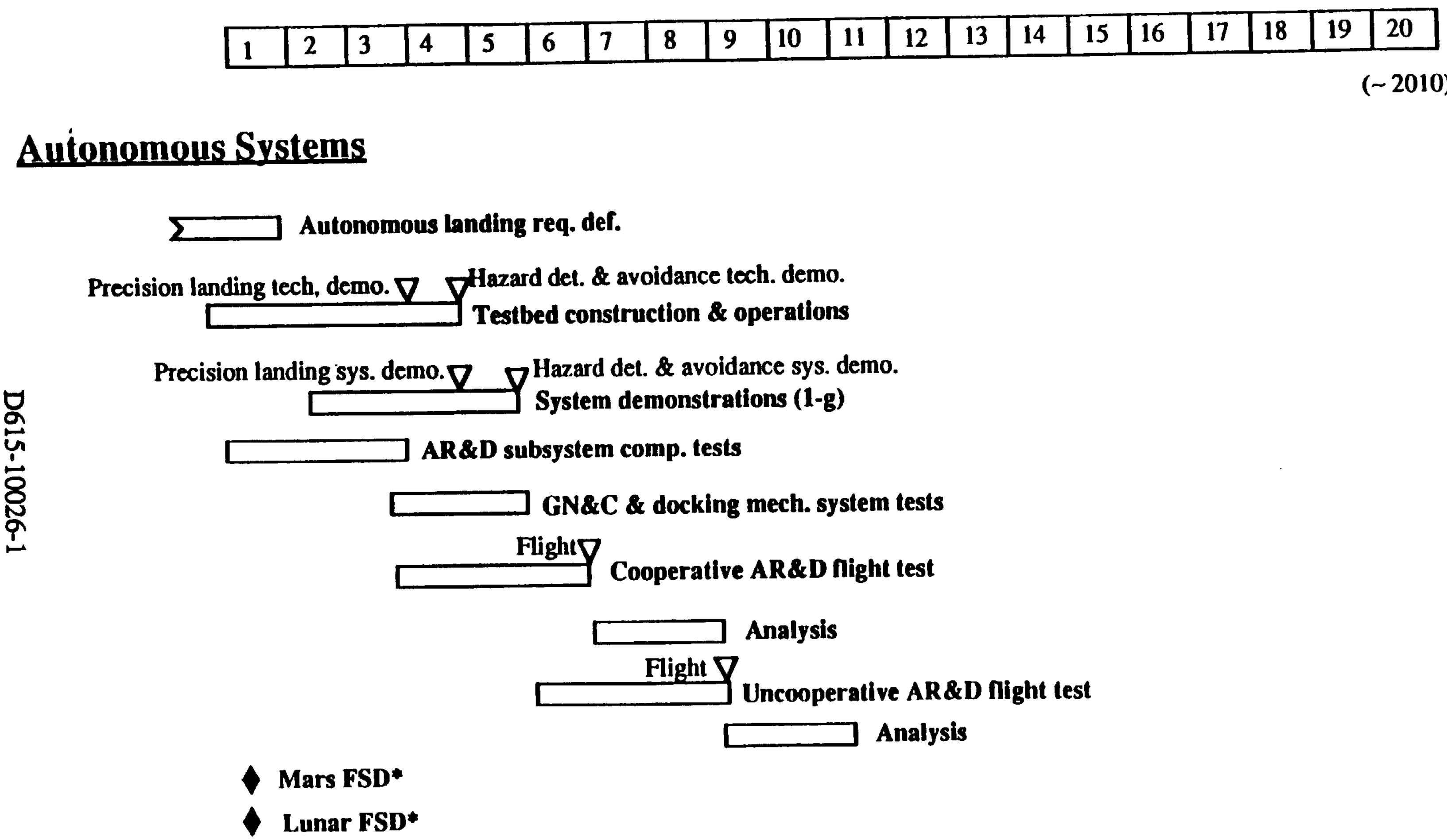

* Technology should not present FSD threatening problems; current technologies adequate for minimum mission.

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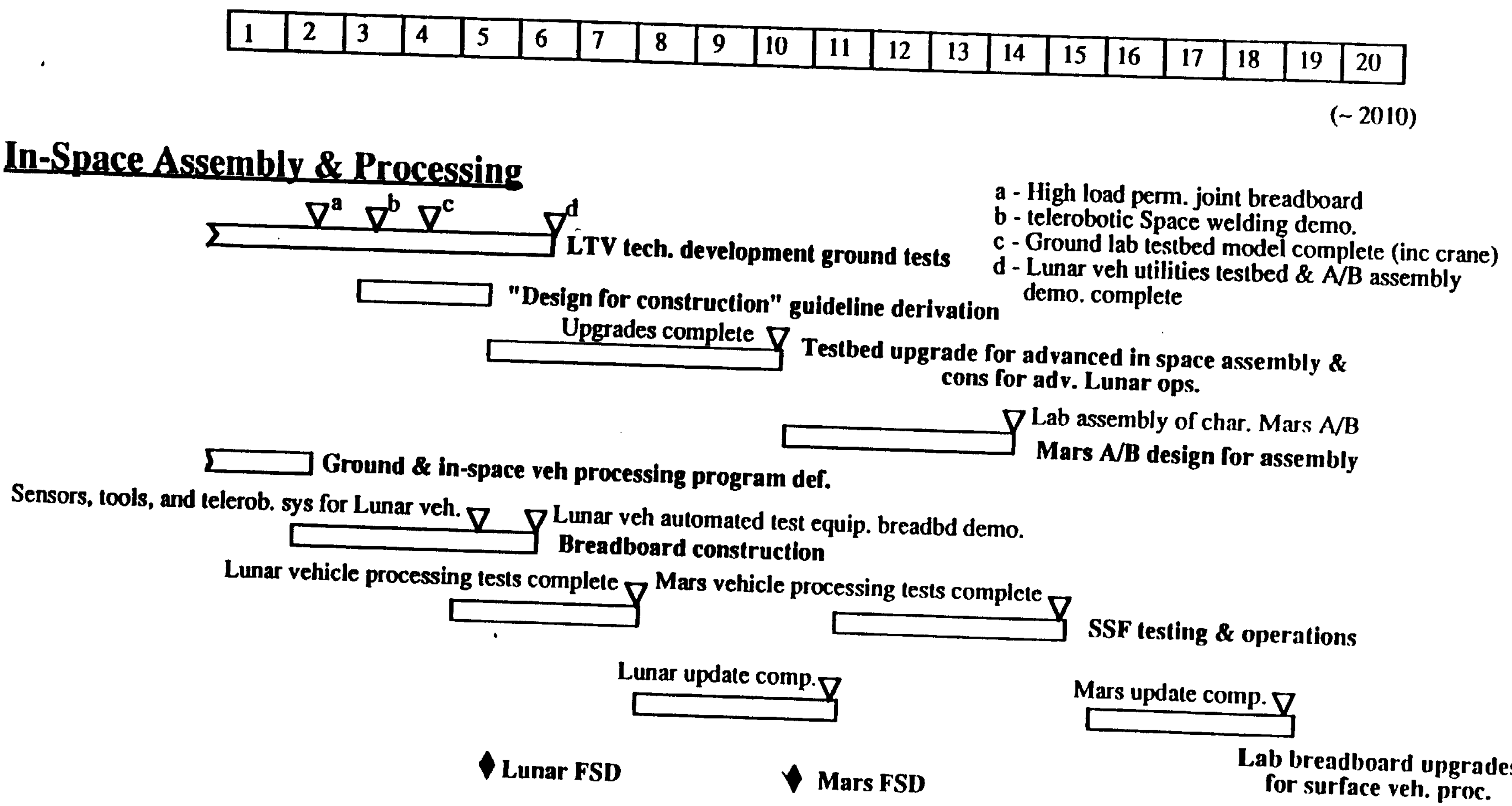
Preliminary SEI Technology Development Schedules (Cont.)



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ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

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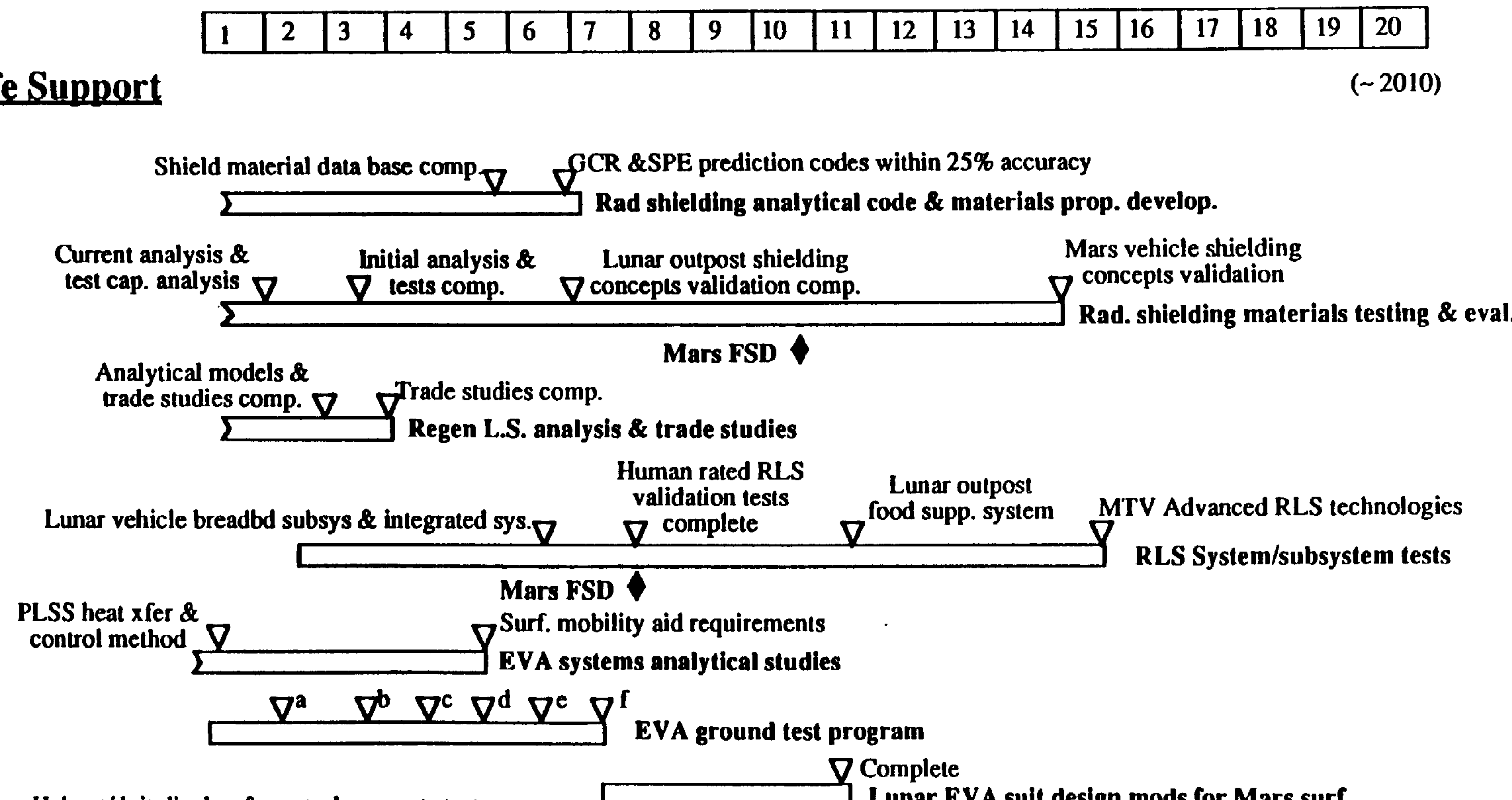
Life Support



a - Helmet/duit display & control concepts tests
b - Lunar surf suit breadbd test
c - Gloves & displays in simul. SSF environment
d - Regen. PLSS breadbd for lunar surf.
e - Verif tests of adv. dexterious gloves & disp.
f - Complete breadbd lunar EVA suit / simulated surf. cond.

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Technology Development Concerns and Schedules - Nuclear Thermal Propulsion (NTP)

Critical technology development issues relating to the reference NTP vehicle are presented in this section. Where applicable, the same charts are also included in the CAB, CAP, NEP, and SEP IP&ED documents. The focus of this section will be to bring out the most important issues relating to the reference NTP vehicle, and to present preliminary technology development schedules for these issues. The issues are presented here in outline form, beginning with the most important, with accompanying schedules wherever possible.

Nuclear Thermal Propulsion Technology Development

The most important area of technology and advanced development for this vehicle option is the development of an integrated nuclear thermal propulsion system. A preliminary schedule for the development of a NTP system for a Mars vehicle is presented. The schedule highlights both the point where a full scale development decision can be made (year 5), and when the first flight article will be available to the vehicle program (year 14). The largest single technology development challenge for the program will probably be test facility design and development. The NERVA program nuclear tests were carried out in a testbed facility open to the atmosphere. Any future test facility must be closed in order to contain the fission products contained in the exhaust gasses. A scrubbing system must be included to remove the fission products from the exhaust gas before it can be released into the atmosphere. This facility may prove to be very costly to build and operate. Nuclear thermal propulsion should offer a shorter development time than the other advanced propulsion options (NEP, SEP), with significantly better performance than the chemical options. The major reactor technology issues are high temperature fuels, efficient frit design, fuel burnup, and nuclear safety issues.

Cryogenic Fluid Management

The large amounts of Hydrogen required for NTP Mars missions increases the importance of technologies development relating to cryogenic fluid management and storage. A preliminary technology schedule is presented for cryogenic fluid system development for Mars mission applications. The cryogenic fluid systems schedule includes Earth-based thermal control and selected component fluid management (tank pressure control, liquid acquisition device effectiveness, etc.) tests, as well as planned flight experiments to carry out system and subsystem development (selected components) and verification/validation tests. Many of the technology issues will be answered during the technology/advanced development work to be carried out for a Lunar program. The major technology obstacles to be overcome by an NTP storage system are in the areas of tankage mass minimization and large scale (relative to Lunar) storage systems development, integration, and orbital/flight operations (fluid transfer, acquisition, etc.).

Vehicle Avionics and Software

Although the technology readiness level of vehicle avionics and software is ahead of many of the other technology areas listed in some respects, the demands on the system in the areas of processing rate, accuracy, autonomous operation, and status/health monitoring will drive technology and advanced development in areas not fully defined at this point. Software requirements cannot be fully determined until the vehicle design is at a more finished stage than the current levels. A preliminary schedule for autonomous systems development is presented. The decision points for full scale development The communications system options can be more fully defined before a final vehicle design is

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Life Support

A reliable, redundant long term life support system will be enabling for future exploration missions. The degree of closure of, and the reliability of the system are the major technology development concerns. Low-g human factors determination will also be an important technology consideration which will drive vehicle design. An integrated schedule of the major areas of the life support technology development task are presented. It includes radiation shielding and materials, regenerative life support, and EVA systems development. As before, the points where Lunar and Mars full scale development decisions can logically be made in the technology program are highlighted.

Aerobraking (low energy)

Low energy aerobraking will offer mission benefits in the areas of decreased demands on the descent propulsion system, and improved crossrange capability. This area presents a variety of issues for technology development including high strength to mass ratio structural materials, high temperature thermal protection systems (although not as high as for high energy aerobraking), avionics, assembly and operations, hypersonic test facilities and computer codes, and Mars atmosphere prediction. High strength structural material options include metal matrix composite, organic matrix composite, and advanced carbon-carbon elements. Other structural considerations include load distribution and attachment of payload for aerocapture, and ETO launch and assembly of large structures. Thermal protection systems issues include low mass ablative and reradiating materials, and structure/TPS integration issues. The aerobrake maneuver will place considerable demands on the vehicle avionics system with the need for real time trajectory analysis, and vehicle guidance and control. The launch and assembly of the large aerobrake structure will present ground and space assembly and ops problems which will require technology and advanced development in both the areas of design and operations. Finally, computational analysis and atmosphere prediction capability will be critical in the development of a man-rated aerobrake for Mars use. A preliminary development schedule for Lunar and Mars aerobrake technology development is presented. It includes the major milestones for both ground and flight testing. The points where a Lunar and Mars full scale development decision can be made are also highlighted on the schedule. It should be noted that this schedule was built with high energy aerobraking in mind, and will possibly be compressed to some degree if only low energy aerobraking is developed.

In-Space Assembly and Processing

The in-space assembly and processing of large space transfer vehicles will present a variety of technology advanced development challenges, particularly for the large LTV and MEV aerobrakes. As shown on the accompanying schedule, extensive ground tests must occur before any orbital work can be initiated. The vehicle designs will be driven to a large degree by the assembly facilities and technologies seen as being available during the vehicle buildup sequence.

Summary

As noted before, many of the identified critical and high leverage technology development issues are common across all of the major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H_2, and possibly O_2 for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique NTP technology issues center around nuclear reactor and engine systems development. Common enhancing technologies include cryogenic refrigeration (lander tanks), O_2-H_2 RCS, advanced in-space assembly techniques, higher Isp cryogenic engines, and advanced structural materials development.

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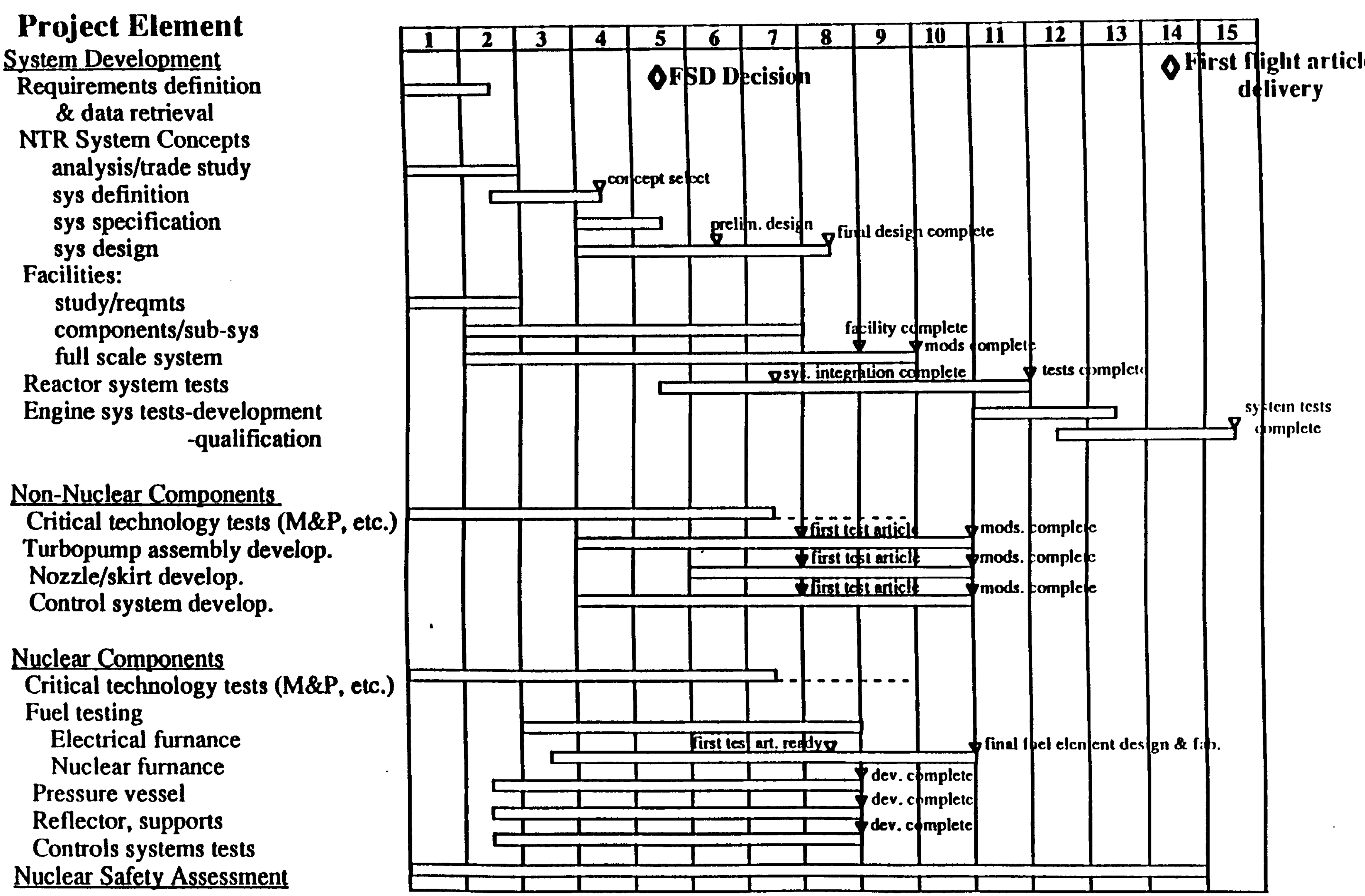
NTR Engine Development Program



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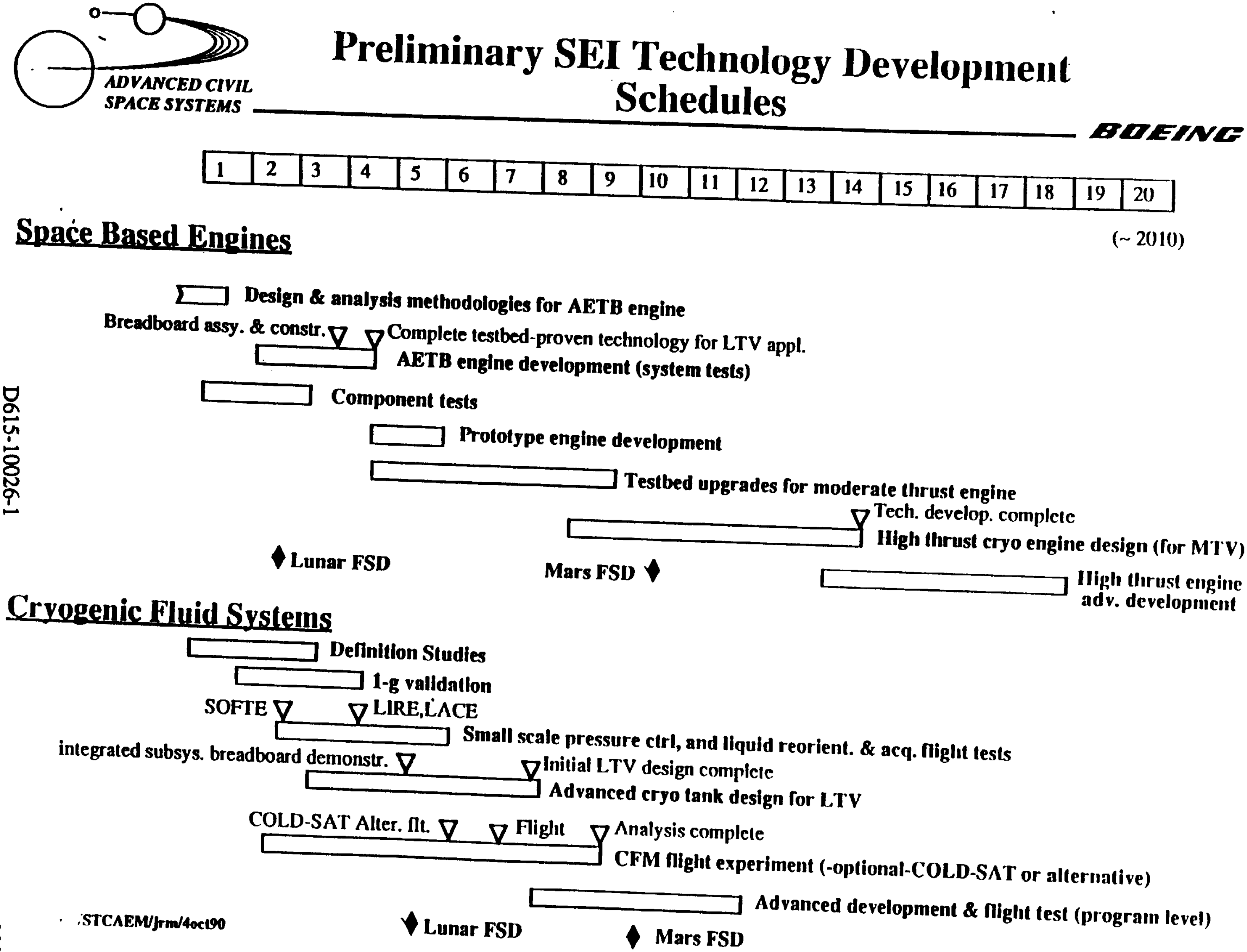
Preliminary SEI Technology Development Schedules



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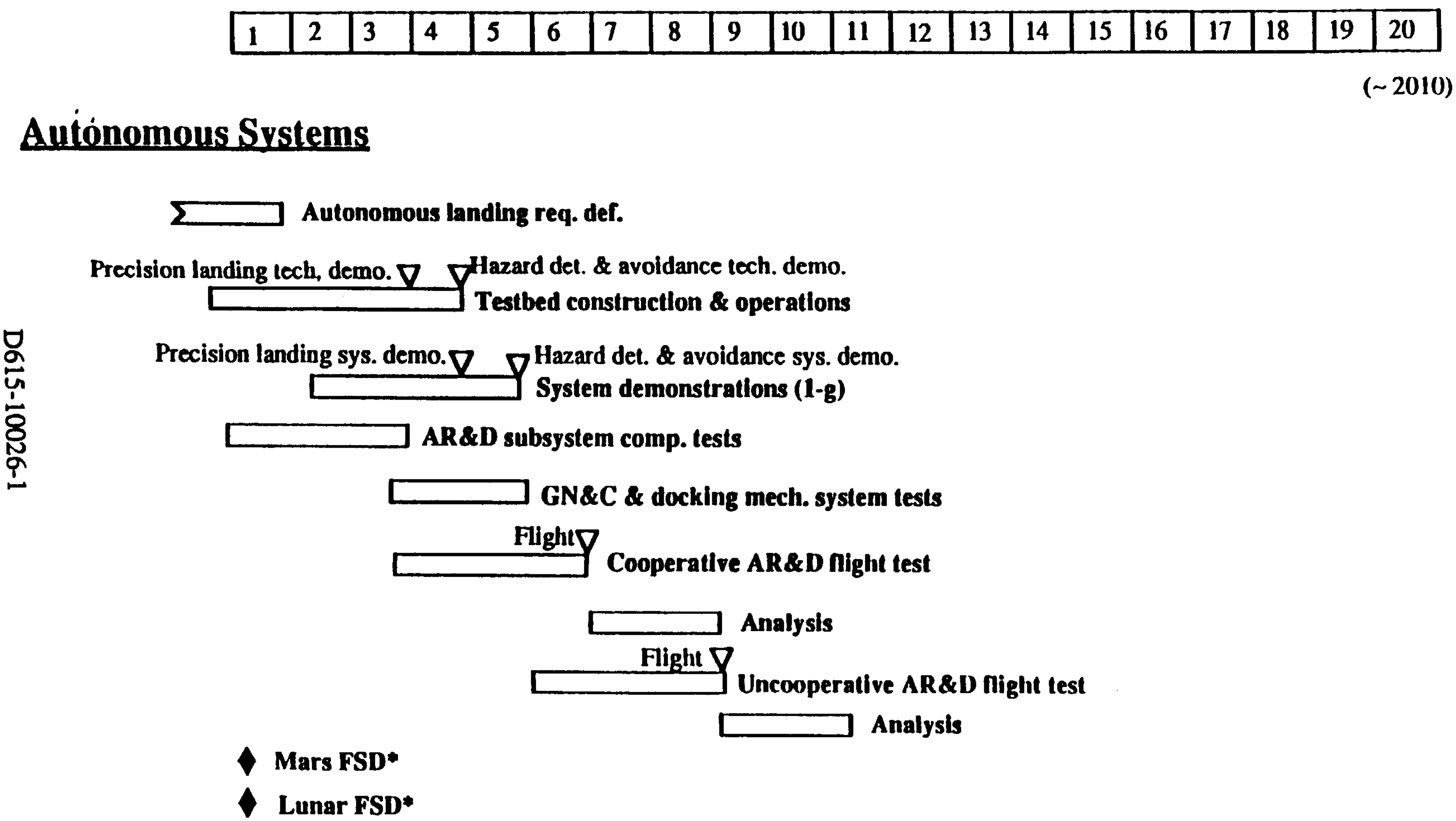
Preliminary SEI Technology Development Schedules (Cont.)



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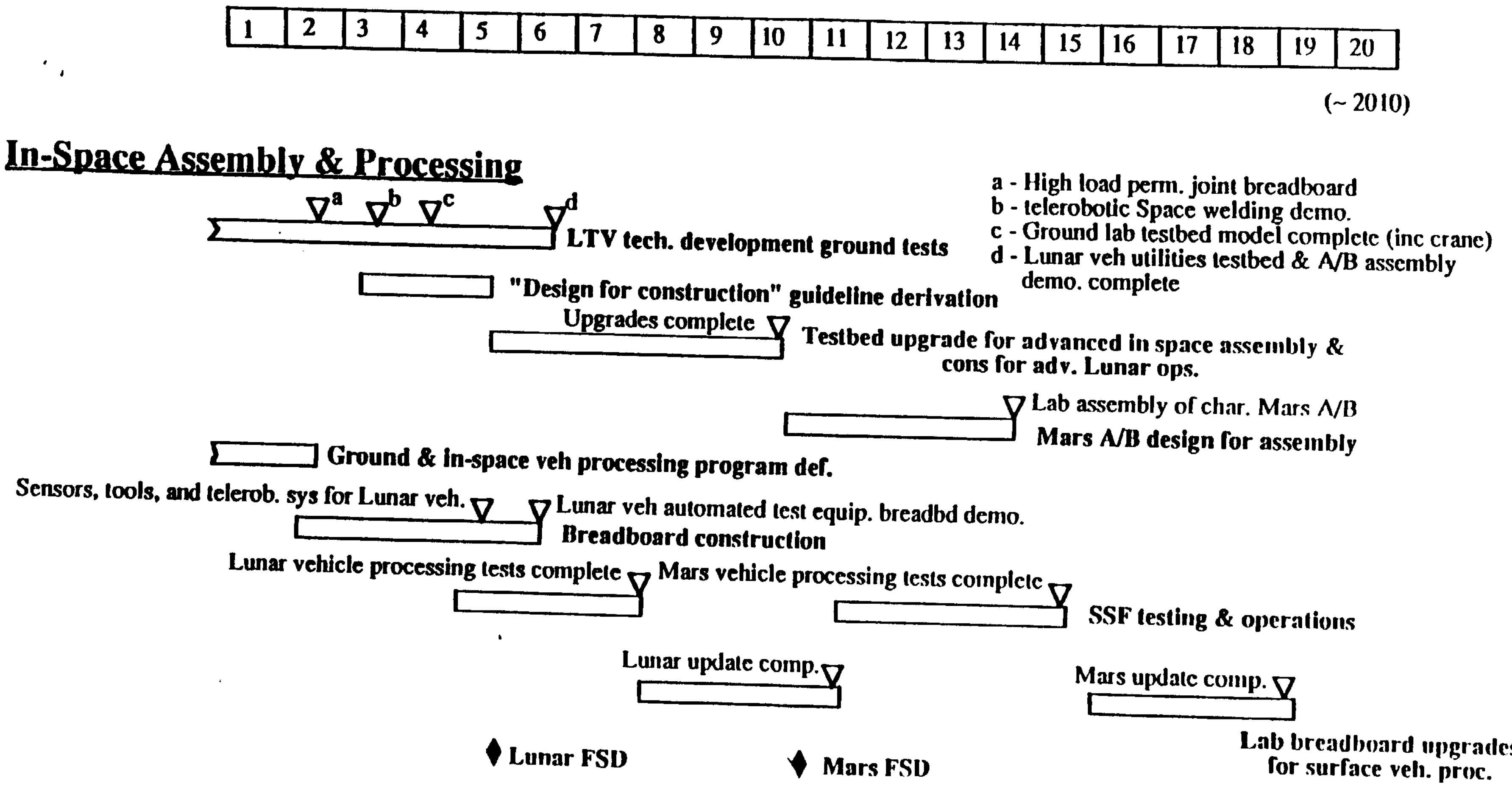
Preliminary SEI Technology Development Schedules (Cont.)



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ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

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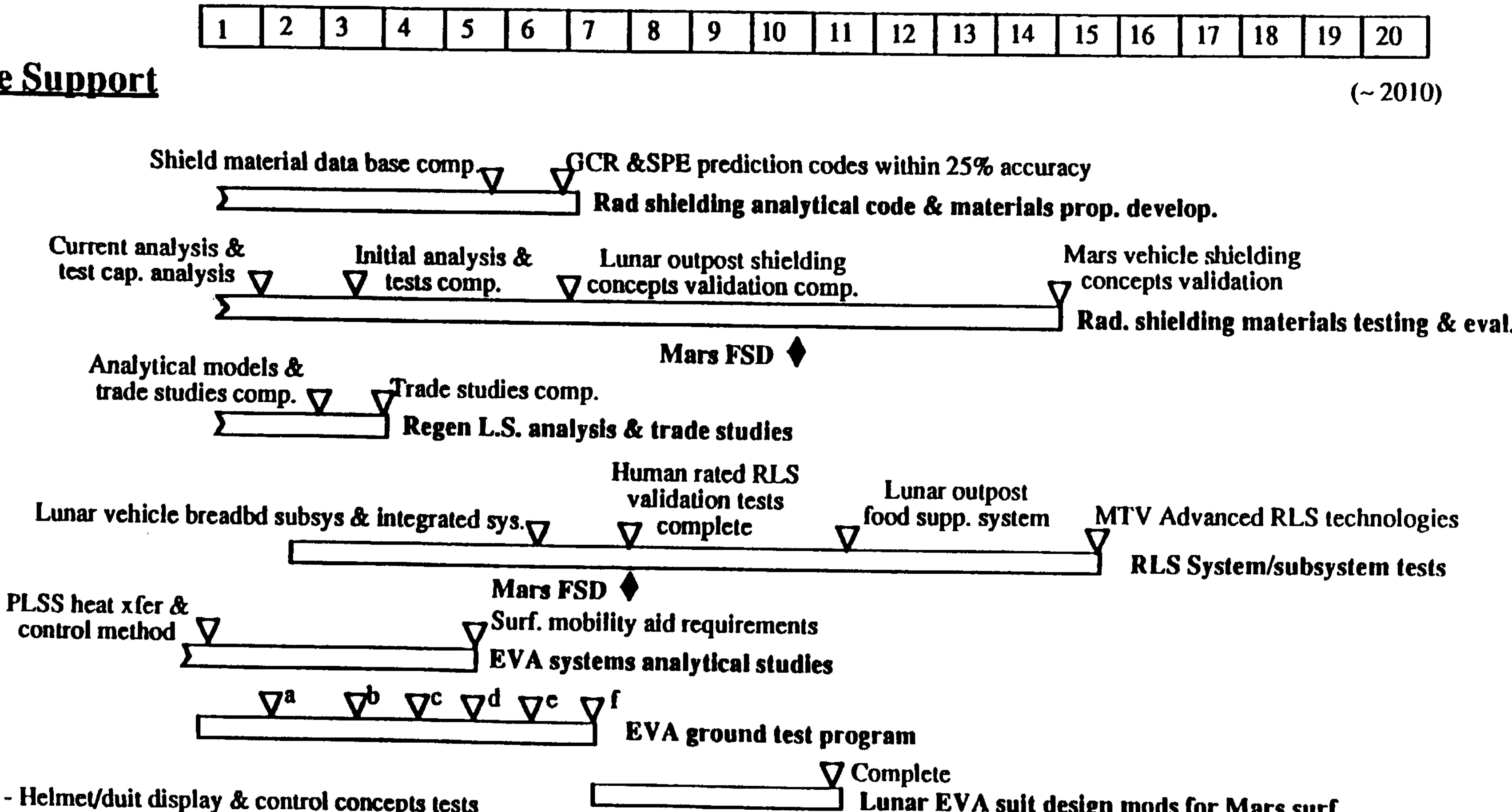
Life Support



a - Helmet/duit display & control concepts tests
b - Lunar surf suit breadbd test
c - Gloves & displays in simul. SSF environment
d - Regen. PLSS breadbd for lunar surf.
e - Verif tests of adv. dexterious gloves & disp.
f - Complete breadbd lunar EVA suit / simulated surf. cond.

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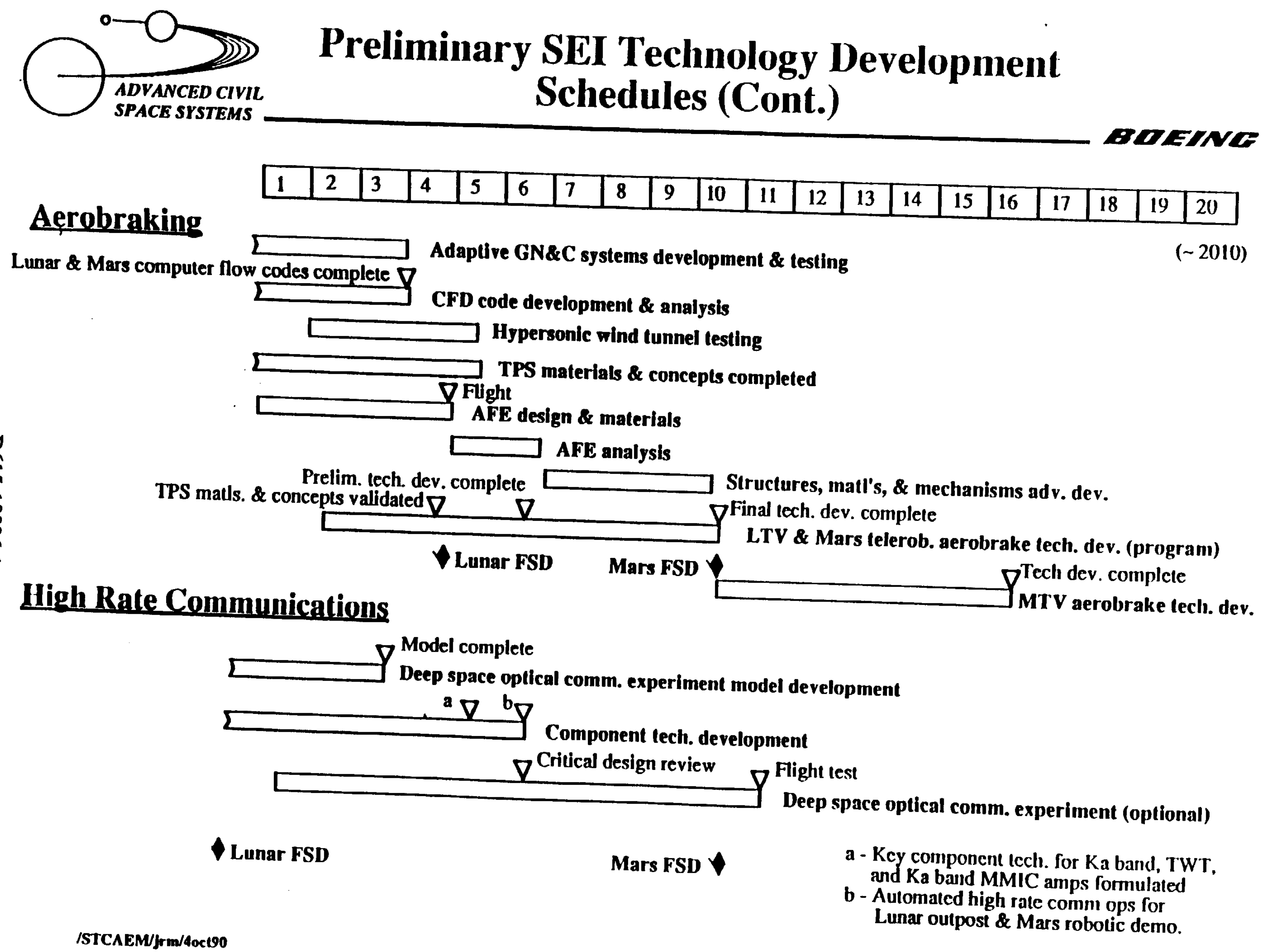
Preliminary SEI Technology Development Schedules (Cont.)



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Technology Development Concerns and Schedules - Solar Electric Propulsion (SEP)

Critical technology development issues relating to the reference SEP vehicle are presented in this section. Where applicable, the same charts are also included in the CAB, CAP, NEP, and SEP IP&ED documents. The focus of this section will be to bring out the most important issues relating to the reference NTR vehicle, and to present preliminary technology development schedules for these issues. The issues are presented here in outline form, beginning with the most important, with accompanying schedules wherever possible.

Solar Power System Technology Development

One of the two most important areas of technology and advanced development for this vehicle option is the development of an integrated solar electric power system. The most important area of development for the SEP option is the design, integration, and life testing of a space qualified multi-megawatt solar power system, consisting of high efficiency solar arrays. Major challenges to be overcome in the achievement of a long life efficient system lie in efficient solar array development, and efficient power processing and delivery systems. Long term life testing must be carried out for the power system in order to verify long term system reliability. A related technology development challenge for the program may be test facility design and development. Solar electric propulsion offers a potential performance which may be superior to the any of the other advanced propulsion options, at the expense of a more costly and lengthy technology and advanced development program.

Electric Propulsion PPU/Thruster Technology Development

The second major area of technology development for the SEP is in large scale electric power processing unit (PPU), and thruster design and development. The power system technology development schedule presented in the NEP IP&ED book includes a timeline for electric thruster design. The development of long life PPU/thruster systems on a larger scale than currently available (MW level thrusters needed) is the major area of concern relating to the SEP concept. Thruster lifetimes on the order of a year or more (continuous) will be required for thrusters on the MW level in scale. Test facilities must be developed which are capable of supporting the long term life tests for these high power level thrusters. Finally, high temperature power processing equipment must be developed to increase system efficiency and reliability.

Life Support

A reliable, redundant long term life support system will be enabling for future exploration missions. The degree of closure of, and the reliability of the system are the major technology development concerns. Low-g human factors determination will also be an important technology consideration which will drive vehicle design. An integrated schedule of the major areas of the life support technology development task are presented. It includes radiation shielding and materials, regenerative life support, and EVA systems development. As before, the points where Lunar and Mars full scale development decisions can logically be made in the technology program are highlighted.

Aerobraking (low energy)

Low energy aerobraking will offer mission benefits in the areas of decreased demands on the descent propulsion system, and improved crossrange capability. This area presents a variety of issues for technology development including high strength to mass ratio structural materials, high temperature thermal protection systems (although not as high as for high energy aerobraking), avionics, assembly and operations, hypersonic test facilities and computer codes, and Mars atmosphere prediction. High strength structural material options include metal matrix composite, organic matrix composite, and advanced carbon-carbon elements. Other structural considerations include load distribution and attachment of payload for aerocapture, and ETO launch and assembly of large structures. Thermal protection systems issues include low mass ablative and reradiating materials, and structure/TPS integration issues. The aerobrake maneuver will place considerable demands on the vehicle avionics system with the need for real time trajectory analysis, and vehicle guidance and control. The launch and assembly of the large aerobrake structure will present ground and space assembly and ops problems which will require technology and advanced development in both the areas of design and operations. Finally, computational analysis and atmosphere prediction capability will be critical in the development of a man-rated aerobrake for Mars use. A preliminary development schedule for Lunar and Mars aerobrake technology development is presented. It includes the major milestones for both ground and flight testing. The points where a Lunar and Mars full scale development decision can be made are also highlighted on the schedule. It should be noted that this schedule was built with high energy aerobraking in mind, and will possibly be compressed to some degree if only low energy aerobraking is developed.

Vehicle Avionics and Software

Although the technology readiness level of vehicle avionics and software is ahead of many of the other technology areas listed in some respects, the demands on the system in the areas of processing rate, accuracy, autonomous operation, and status/health monitoring will drive technology and advanced development in areas not fully defined at this point. Software requirements cannot be fully determined until the vehicle design is at a more finished stage than the current levels. A preliminary schedule for autonomous systems development is presented. The decision points for full scale development The communications system options can be more fully defined before a final vehicle design is produced, however. A technology development schedule for advanced communications is presented. The SEP vehicle may not place the same level of demand on the avionics system in the area of trajectory analysis, but will likely place more demands on the system in the areas of status and health monitoring, fault diagnosis, and correction.

In-Space Assembly and Processing

The in-space assembly and processing of large space transfer vehicles will present a variety of technology advanced development challenges, particularly for the large LTV and MEV aerobrakes, and SEP vehicle. The large solar array structure, along with the large amount of wiring and electrical connections will present a variety of challenges in technology development (e.g. in-space welding), and assembly operations (e.g. robotics). As shown on the accompanying schedule, extensive ground tests must occur before any orbital work can be initiated. The vehicle designs will be driven to a large degree by the assembly facilities and technologies seen as being available during the vehicle buildup sequence. It should be noted that the schedule was not developed specifically for an NEP vehicle. Advances derived from this development process along with flight experience in earlier missions leading up to this evolutionary scenario could possibly accelerate the development plan considerably.

Cryogenic Fluid Management

The level of concern for technology development in the areas of cryogenic fluid management and storage will not be as for electric propulsion vehicles as for the high thrust systems, although many of the areas still remain important for the SEP vehicle. The Argon (or Zenon) propellant utilized for the electric propulsion system will be in a cryogenic liquid state, and will require long term storage and management technology levels similar to those

GENERAL DYNAMICS

Space Systems Division

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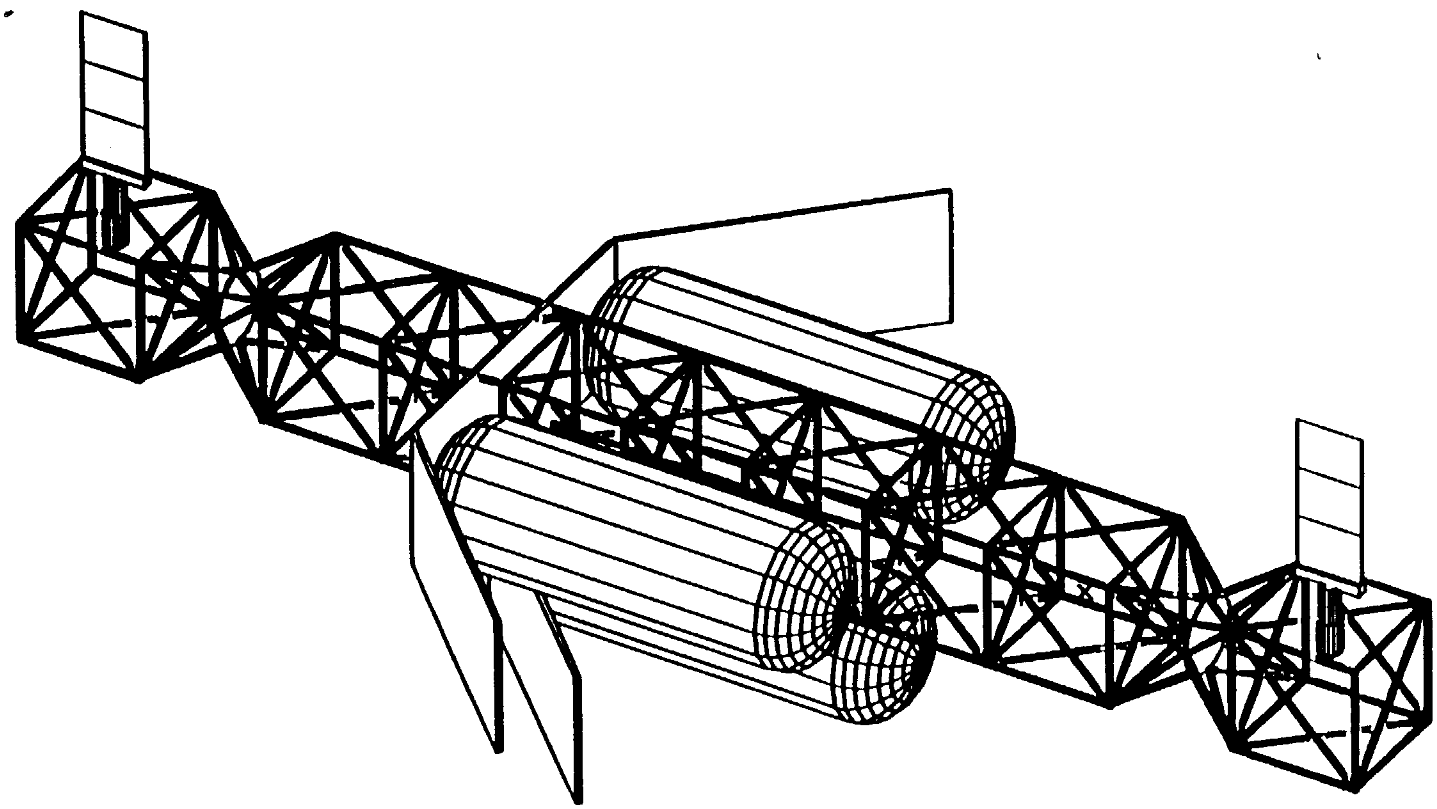
Depot Concept for Support of CAP Vehicle

GENERAL DYNAMICS
Space Systems Division

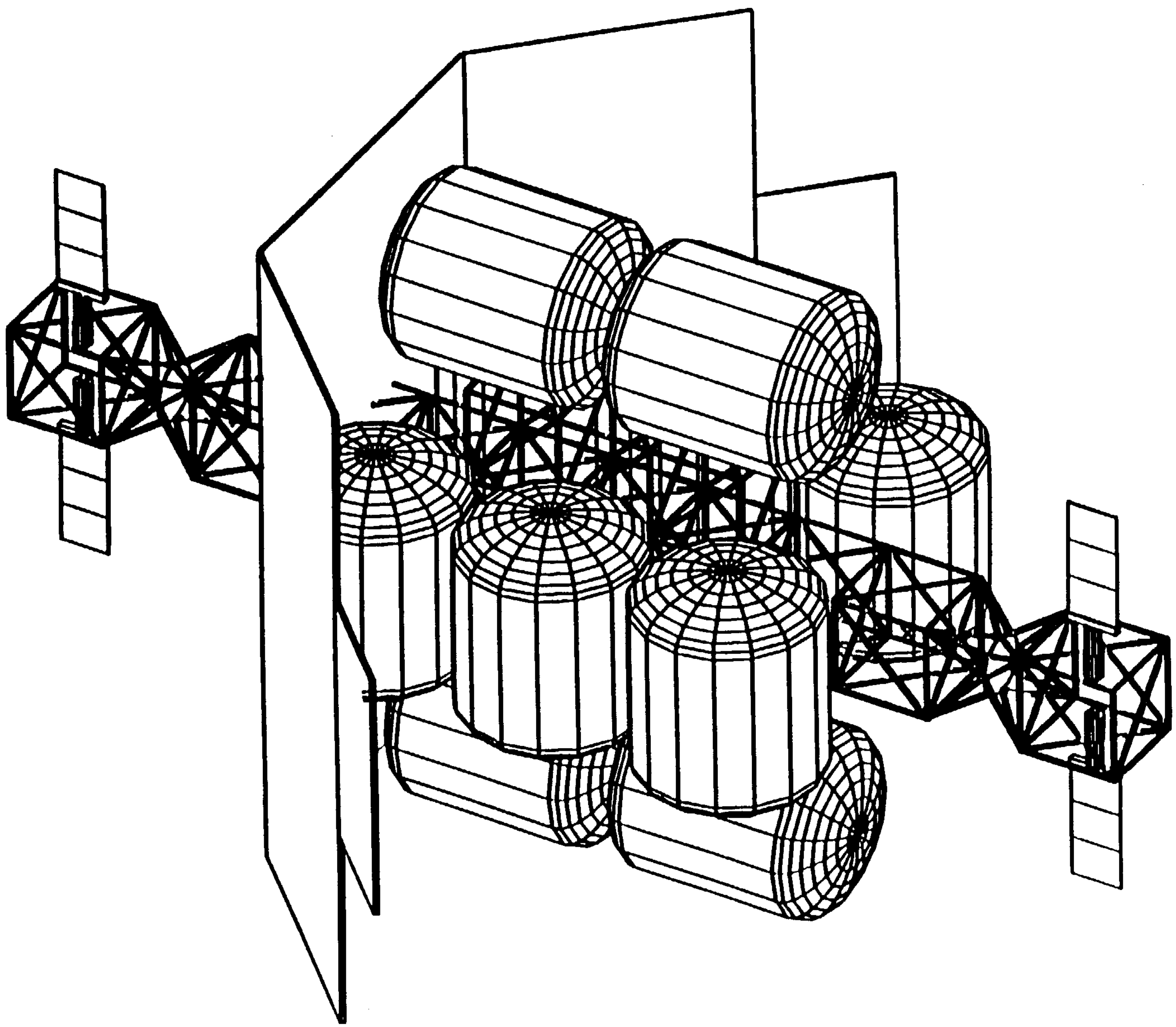


Depot Concept for Support of NTR Vehicle

Passive thermal controlled, vented zero-g transfer depot for support of NTR vehicle.

Depot Concept for Support of NTR Vehicle

GENERAL DYNAMICS
Space Systems Division



ET Shielded Depot Configuration

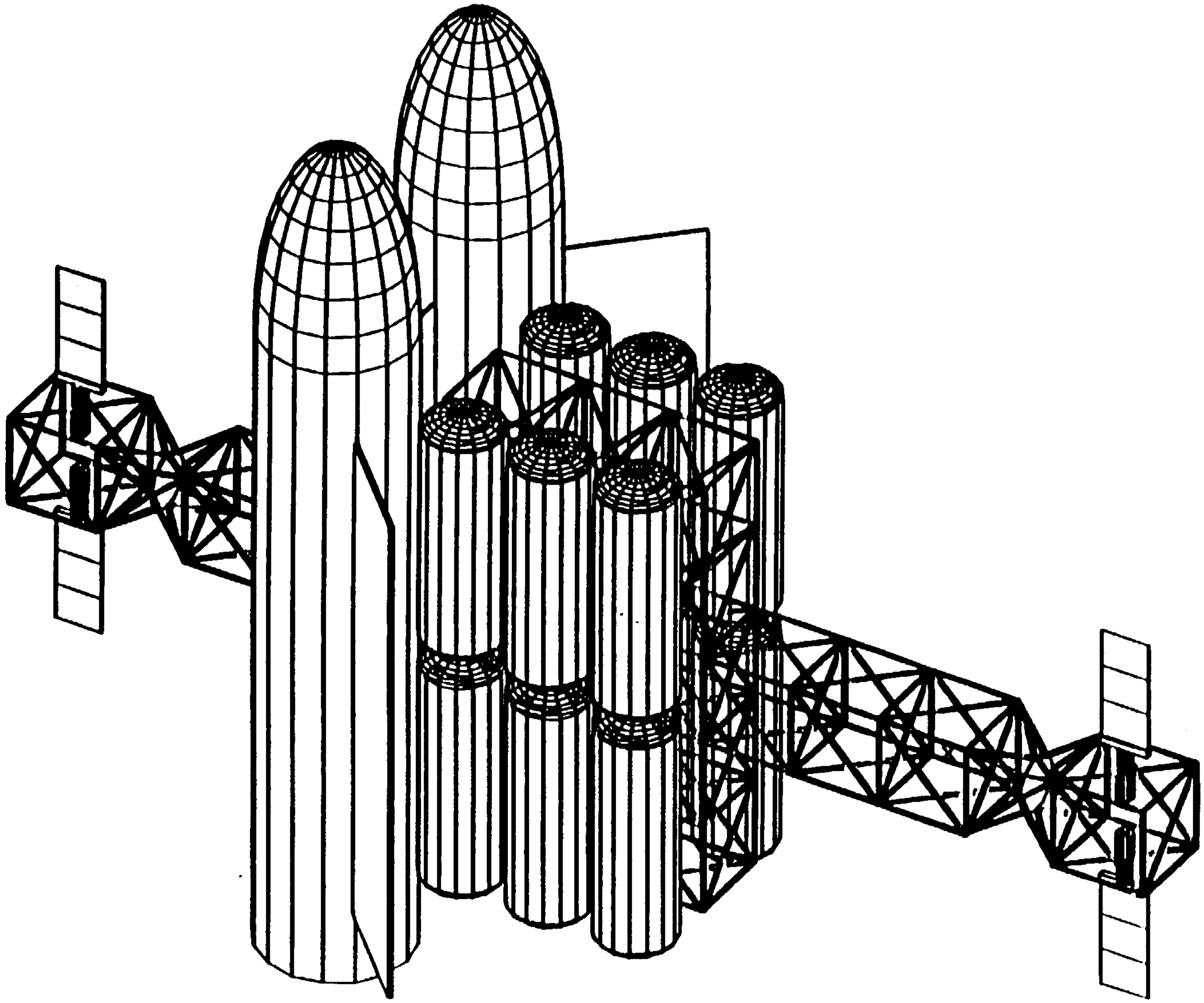
Alternative configuration showing possible utiliization of STS External Tank as source of shielding for depot.

ET carries 98% of required velocity for orbital insertion.

Large diameter, semi-monocoque construction offers adequate protection for tanksets.

ET Shielded Depot Configuration

GENERAL DYNAMICS
Space Systems Division



Gravity Gradient Stabilized Depot Configuration

Alternative configuration shows possible gravity gradient stabilized method of inducing artificial gravity for propellant feed.

Unequal balance of forces causes settling of propellant at ends of truss:

- Centripetal force overcomes gravity, at space end
- Gravity overcomes centripetal force at earth end

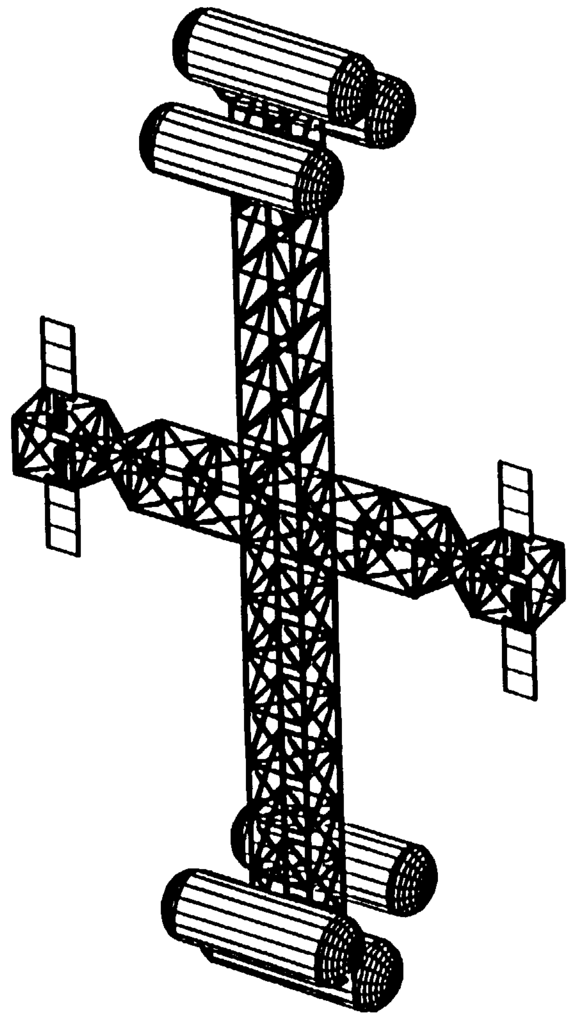
Minimum required truss length for sufficient gravity to sufficiently overcome surface tension:

> 70 meters

Acceleration felt by fluid:

$\approx .4 \times 10e\text{-}04$ g

Gravity Gradient Stabilized Depot Configuration



CAP Depot Configuration Propellant Requirements

CAP missions require off-loading of propellant from vehicle tanks to fully utilize mass and volume capability of launch vehicle.

Top off propellant is fuel which has been off-loaded due to mass constraints on the launch vehicle capacity.

Boiloff and transfer losses were calculated for propellants based on their launch sequence and duration on orbit.

Sum of these amounts represents required capacity of depot.

High boiloff amounts could be greatly reduced by increasing insulation, reducing launch centers to reduce cumulative duration spent on orbit prior to launch.

CAP Depot Configuration Propellant Requirements

GENERAL DYNAMICS
Space Systems Division

All Missions

2 x 52 t Off-loaded propellant	104.0 tons
7 t Off-loaded propellant	7.0
Top off	90.0
Boiloff/Transfer	119.5
Total	320.5

- As high boil-off and transfer rates are cause for launch of an extra tank to depot, additional insulation or closer launch centers are required to keep down losses due to boiloff

NTR Depot Configuration Propellant Requirements

The NTR missions have an additional component to support--propellant for LTV.

LTV propellant required for rendezvous with returning NTR vehicle. Propellant for orbit circularization is transferred to the vehicle, after which it conducts its burn and returns to LEO.

Depot sized by initial demand for Missions 1-3--seven hydrogen tanks and one oxygen tank.

Missions 5-6 require additional propellant to support heavier payload--two additional hydrogen tanks needed to give depot required capacity.

NTR Depot Configuration Propellant Requirements

GENERAL DYNAMICS
Space Systems Division

Mission #1

Aft Tank	54.6 tons
Top off	90.0
LTV	89.0
Boiloff/Trans.	34.0
Total	**267.6**

Mission #2

Aft Tank	60.0 tons
Top off	90.0
LTV	89.0
Boiloff/Trans.	33.2
Total	**272.2**

Mission #3

Aft Tank	59.2 tons
Top off	90.0
LTV	89.0
Boiloff/Trans.	34.0
Total	**272.2**

Mission #4

Aft Tank	60.0 tons
Top off	120.0
LTV	89.0
Boiloff/Trans.	34.0
Total	**303.0**

Mission #5

Aft Tank	60.0 tons
Top off	210.0
LTV	89.0
Boiloff/Trans.	53.4
Total	**412.4**

Mission #6

Aft Tank	60.0 tons
Top off	210.0
LTV	89.0
Boiloff/Trans.	53.4
Total	**412.4**

- **Depot initially sized for Missions 1-3 capacity; additional tanks added during subsequent missions to add depot capacity to support Missions 5-6.**

Solar Panel / Power Parameters

Solar panels are deployable planar arrays using GaAs cells baselined with 18% efficiency.

Sun tracking mechanism employs alpha and beta joints.

H_2 tanks require 400 W each; H_2 / O_2 tanks require 660 W each.

Longer shadow time for NTR configuration requires additional oversizing of solar panels.

CAP configuration locates panels on space side of truss to minimize shadowing by the shield.

Solar Panel / Power Parameters

GENERAL DYNAMICS
Space Systems Division

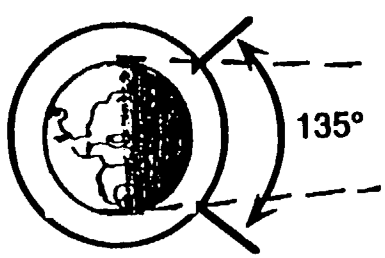
Common Issues

- Eclipse Time ≈ 33% of Period

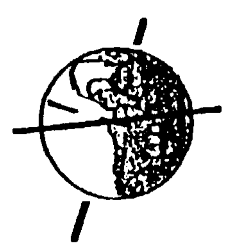
- Baselined GaAs Cells
  - 18 % Efficiency
  - 80 % Absorbtivity
  - 82 % Emissivity

- 400 W / tank baselined for all passive H2 tanks

- 660 W / tank baselined for a ll passive H2 / O2 tanks

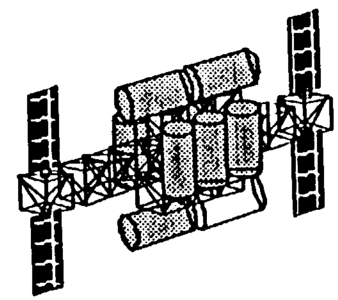


Pole View



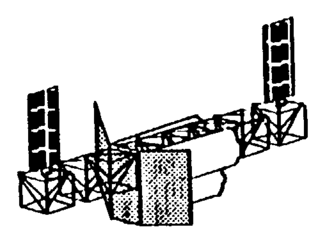
Ecliptic View

NTR Configuration



Shadowed by Shielding 35% of Period
64.5 m2 ≈ 4 kW Steady State

Cryo All Propulsive Configuration



Shadowed by Shielding 6% of Period
33.0 m2 ≈ 2 kW Steady State

Shielding Configuration

Shielding concept incorporates:

Aluminum outer shield--Absorbs bulk of initial impact energy from projectile; deflects penetrating particles and ejecta from direct path to tank, spreading secondary impact over larger area.

MLI--Absorbs remaining energy from projectile and ejecta, leaving tank wall free from shielding system.

Aluminum inner sheet--Gives structural stability to shield, additional protection.

Standoff--Allows projectiles and ejecta room to disperse over larger area, minimizing energy concentration of initial impact.

Winged approach to shielding accomplishes:

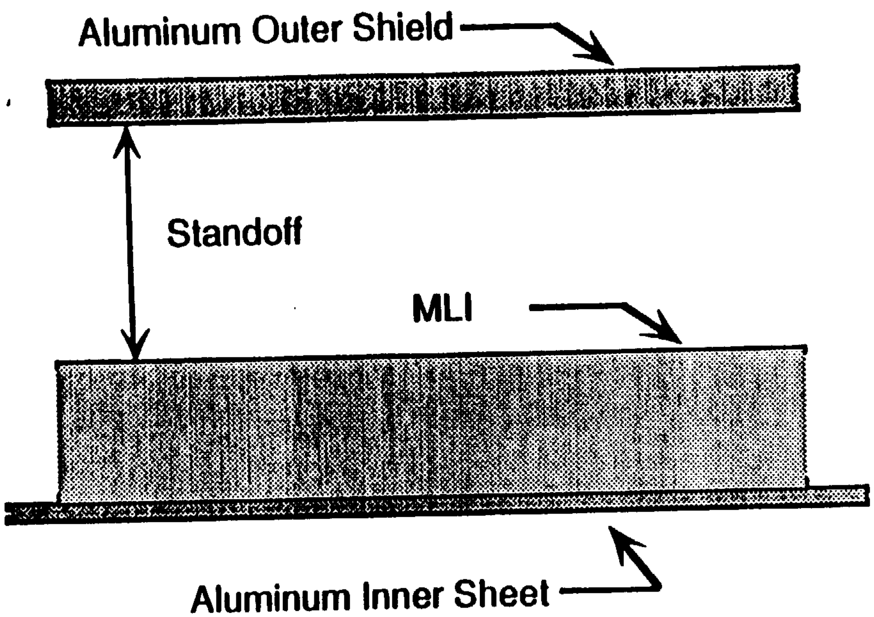
-More economic utilization of weight over smaller surface area than achievable by wrappin tanks.

-Weight reduction; no recurring weight penalty for subsequent tank launches.

45° angled wings cover forward facing surfaces of tanks where orbital debris impacts are most likely.

Shielding Configuration

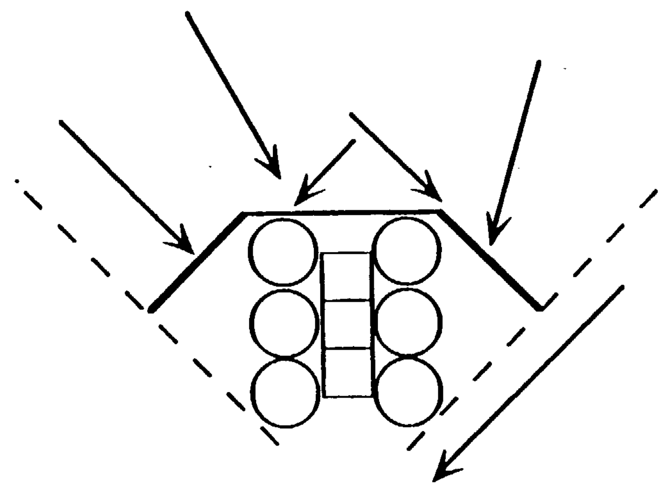
GENERAL DYNAMICS
Space Systems Division



- The winged approach to non-integral shielding helps minimize:

  Surface area--more economic means of covering tanking faces than individual wrapping

  Weight--makes shielding a one time launch item, no recurring penalty for subsequent tank launches



Top View

- Several sources have promoted use of a composite shield employing an aluminum outer debris shield and MLI in conjunction with a standoff

Operations / Integration Issues

MTV and depot manifested as separate missions:

- -Sufficient variety in MTV component list to fully utilize mass and volume capability of launch vehicle.
- -MTV and depot launch / assembly operations should not be concurrent to avoid impacting one another.
- -Assembly needs platform (STS or SSF) for initial truss deployment to minimize EVA demands.

Telemetry must exist to remotely monitor depot systems, as no manned habitat, crew safe haven, or provision for life support will be part of depot.

RMS needed for delivery of tanksets from LTV or other cargo transfer vehicle.

Reboost / deboost operations will be monitored remotely; resupply missions will coincide with reboost phase to facilitate propellant delivery, minimize boiloff.

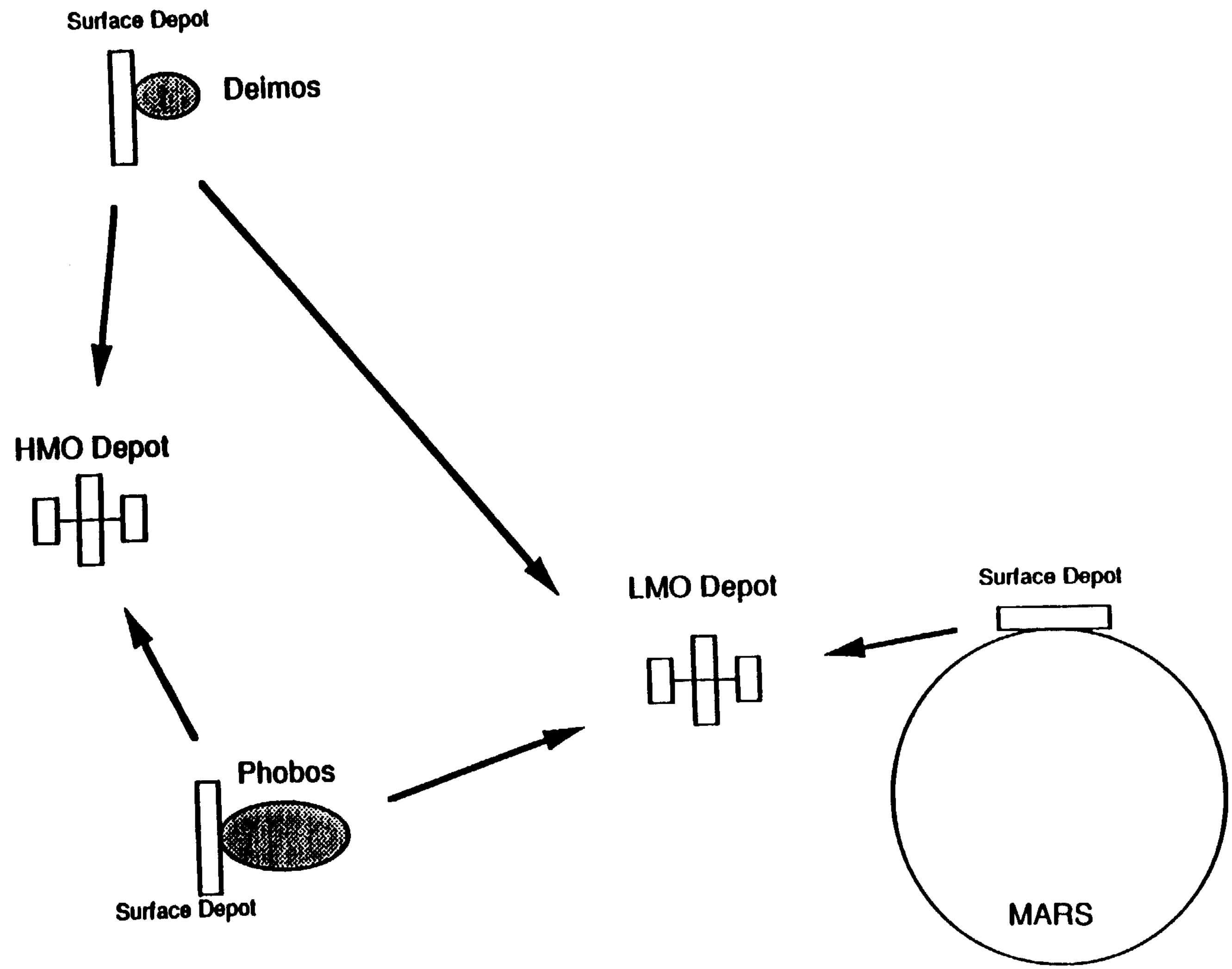
Operations / Integration Issues

- MTV and Depot were manifested as separate ETO missions:
  - Sufficient flexibility existed in MTV component list to allow either full volume or mass capacity usage in most cases w/o need to infuse depot components or separate H_2 / O_2 tanks in CAP scenario
  - Depot launch / assembly should occur prior to inception of assembly on MTV to avoid impacting launch schedule, assembly crew load.
  - Assembly would benefit from an initial platform (either STS or SSF) from which to deploy a truss
- Truss structure shall incorporate deployable design to minimize EVA demads.
- Telemetry / communications must exist to allow man-tended, remote monitoring capability of depot systems.
- No manned habitat or crew safe haven will be part of depot configuration. Life support shall be the sole responsibility of the crew transfer vehicle.
- RMS required at depot for transfer of cryogenic storage tanks from LTV or other ferrying system used to transport, dock cryogenic storage tanks.
- Reboost / deboost operations will be monitored remotely.
- Reboost / deboost engines feed from tank residuals, necessitating resupply mission timing to coincide with reboost phase.

Mars Depot Architecture Concept Options

Many Mars system depot architecture options are possible to support a large-scale Mars exploration program. Wherever primary propellant production sites are located - Phobos, Deimos, or Mars surface - a surface propellant depot capable of storing and protecting cryogens against the ambient environmental conditions will be required. At least two orbital locations immediately suggest themselves: low Mars orbit (LMO) and high Mars orbit (HMO); an HMO depot would have orbital altitudes comparable to the moons. An LMO depot would be a logical alternative if propellants were originating on Mars, although it could also be supported from Phobos and Deimos. Likewise, propellants originating on the moons would suggest the use of an HMO facility. The most attractive option will be determined by specific Mars mission and program scenarios.

Mars Depot Architecture Concept Options

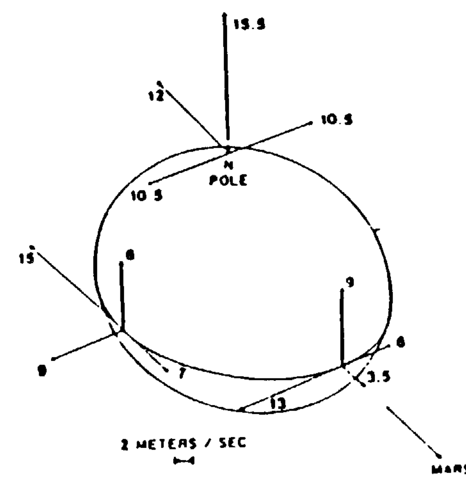


Mars Environmental Issues

Operating near and on Mars means encountering an entirely new set of orbital, thermal, gravitational, atmospheric, and surface environments that can be crucial to the operation of a Mars orbital propellant depot. Some of these parameters are well-known but others have not yet been well-specified. The average Bond albedo (0.16) results in the planet having much lower daytime and nighttime blackbody surface temperatures, so objects in Mars orbit receive less thermal radiation from Mars than they would a comparable orbits around Earth. The general meteoroid environment near Mars is well-known from several spacecraft missions over the last three decades, however, there are good theoretical reasons to suspect that belts of dusts may exist in orbits near the satellite moons (particularly Phobos). If operations are contemplated near Phobos and Deimos, their complex local gravity field must be better understood. Operating on the Mars surface itself is a complex, potentially dangerous, highly challenging proposition. The 24 hour diurnal cycle may drive the use of nuclear power on the surface.

Mars Environmental Issues

- Thermal Environments
  - Subsolar BB T=270 K
  - Darkside BB T=190 K
- Meteoroid Environment
  - Nominal
  - Satellite Dust Belts
- Satellite Gravity Environments
  - Phobos
  - Deimos
- Mars Surface Environments
  - Diurnal/Seasonal T/insolation variations
  - Winds/Dust Storms



Phobos Escape Velocities

Mars Depot Location Option Summary

Five general locations (and 9 specific sites) suggest themselves as potential locations for Mars propellant depots; they are: Mars orbit (free space), Phobos, Deimos, Mars' surface, and the Sun-Mars libration (L) points. Each location is characterized by its environmental attributes, nearest location of primary propellant production, and relation to the anticipated Mars system infrastructure and exploration program.

High circular orbits (i.e. those with semimajor axes comparable to the moons) are accessible to Phobos and Deimos and are far removed from the Mars thermal radiation source. Low Mars orbits are convenient to surface operations (the ultimate focus of any Mars program) and the most likely location of primary propellant production: the Mars surface. An LMO depot may be the most likely location for an early depot within the Mars system.

The Martian moons provide many strategic and operational advantages associated with the exploration of Mars. Their low gravity fields and potential for in situ propellant production make them potentially important early targets for exploration and utilization.

The most likely site for some type of primary propellant production is on the surface of Mars using the regolith and/or the atmosphere. The 1/3 g surface gravity field should make fluid and human operations quite normal. However, the Martian surface environment - particularly its thermal radiation - will provide the most challenges for making and storing cryogenics.

Mars Depot Location Option Summary

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Location		ADVANTAGES	DRAWBACKS
MARS ORBIT	High Ellip/Inc	Lowest capture delta-v; mod thermal loads Low plane change delta-v; zero-g transfer	Relatively high delta-v to Mars' surface, LMO circ, and Ph/D
	High Circ/Eq	Low delta-v for Ph/D transfer; low thermal loads; zero-g fluid transfer	Higher delta-v to Mars surface and LMO; Possible dust belts
	Low Circ	Convenient to LMO and surface propellants Zero-g fluid transfer	High thermal loads; High delta-v for escape, capture or Ph/D
PHOBOS	Surface	Probable propellant production site; Thermal shield from Mars; milli-g fluid ops	Heat conduction an issue; Dust env.; milli-g surface ops; Delta-v to LMO
	Vicinity	Convenient to propellant production site Potential Mars thermal shielding	Complex local gravity field; Higher delta-v to LMO & surface
DEIMOS	Surface	Same as Phobos surface; Most convenient moon for Mars capture/escape (MTV ops)	Same as Phobos surface
	Vicinity	Same as Phobos vicinity; Stable orbits around satellites (inc. Phobos) exist	Same as Phobos (although local gravity less complex than Phobos)
MARS SURFACE		Probable propellant production sites (e.g. regolith and/or atmosphere); 1/3 g fluid & surface ops; Simpler facility requirements than those of orbital depots	Highest thermal loads; Surface environment (e.g. dust storms) a problem; Propellants possibly stored as H2O and then split & liq
L POINTS		Convenient for MTV operations; Very low thermal loads; Possibly useful in some Mars exploration scenarios	Long travel times to LMO & Mars; Far from Mars system propellant production sites and infrastructure

LEO-Mars Depot Commonalities - Preliminary Assessment

System, technology, and operational similarities between depots planned for potential Martian use and those contemplated for LEO are important issues. If significant commonalities exist between LEO and Mars systems, and if a mission indeed exists for a Mars depot, then cost and schedule savings might be obtained. This top-level look at commonalities suggests that, in some locations near Mars, the LEO system might be overdesigned and susceptible to significant transference. Conversely, it is also possible that the identification of both an important mission for a depot at Mars and significant LEO-Mars depot commonalities, might influence planners to recommend that such a depot be built even if the case for a LEO depot is only marginal.

Our baseline thermal management system is all passive and vented. A similar system might be used in either LMO or HMO where the boiloff due to planetary thermal radiation will be less than in LEO. However, particularly in LMO, a Martian depot might benefit from the capability to avoid or reliquefy any boiloff due to the scarcity of Martian propellants and/or because of the presence of science instruments or manned operations in the vicinity of the depot. While we envision that tank changeout will be utilized in LEO, this requires a powerful RMS such as that at SSF and it is unclear whether such a system will be available near Mars. In it absence, fluid transfer would be the preferred technique. Nuclear power systems seem preferable on planetary and satellite surfaces because of their diurnal cycles and surface/atmospheric environments.

GENERAL DYNAMICS
Space Systems Division

LEO-Mars Depot Commonalities - Preliminary Assessment

SUBSYSTEMS/ OPERATIONS	LEO	LMO	HMO	MARS SURF	PHOBOS/ DEIMOS
Thermal Management	All passive/ vented	Refrig/ Relique (?)	All passive/ vented	Refrig/ Relique	All passive/ vented
Fluid Management Acquisition	Zero-g	Zero-g	Zero-g	1/3 g	Milli-g
Transfer	Tank Transfer	Fluid Transfer	Fluid Transfer	Either/Both	Either/Both
Power Supply	Solar	Solar or Nuclear	Solar	Nuclear	Nuclear (?)
Debris/ Meteoroid Protection	Both	No Debris	No Debris	Surface Env. Protection	Meteoroid & Dust
Communication/ Tracking	TDRSS Constell.	Mars TDRSS	Mars TDRSS	NA	NA
Manned Operations	Man-tended	Man-tended	Man-tended	Manned	Manned or Man-tended
Orbit Maintenance	Reboost (near SSF)	Minimal Reboost	NA	NA	NA
Other Depot Functions	None	Science; Vehicle Serv.	Science; Vehicle Serv.	NA	Science

Issues for Further Study

The Minimum Science Program and Full Science Scenario are only two of the many scenarios that can be envisioned for human Mars exploration programs. It is unlikely that a Mars propellant depot would be required for the former, but any science, resource, or settlement scenario that involves significant operations near and/or on Mars will necessitate the use of in situ resources, in particular the in situ production of propellants. A depot can be a logical step in many of them. There is the suggestion from this analysis that a Mars depot should be examined further.

Issues for Further Study

- Broader spectrum of Mars Exploration Options
- Mars depot architecture options
  - Surface & Satellite Propellant Production
  - Depot locations
- Commonality Issues and Strategies
  - Depots in: LEO, Mars orbital and surface, Phobos/Deimos
  - Influence of commonality on depot architectures
- New Technology Requirements

Key Potential Vehicle Safety Hazards

The safety hazards present for vehicle operations on-orbit are identified and recommendations for their potential abatement are presented.

Key Potential Vehicle Safety Hazards

HAZARDS	RECOMMENDATIONS
• TANKS BECOMING PROPULSIVE	
- Meteoroid/Debris impact & puncture	Provide debris shield; Tank jettison capability
- Rupture due to overpressurization	Automated monitoring and release of pressure
- Ignition of H/O (Requires H/O mixing some pressurization and energy)	Separate H & O tanks; Ambient space pressure too low for combustion
• EVA HAZARDS	
- Leaks cause EVA suit contact with LO2	Monitor & avoid leaks; Minimize EVAs
- Thruster plume impingement on EVA crew	Coordinate EVA & reboost times; Avoid thrusters
• DEPOT OPERATIONS - LARGE MASS TRANSFER	
- RMS requirement for tank changeout - Mechanical/Fluid interfaces for fluid transfer	Minimize number of tank changeouts and fluid transfers
• NUCLEAR VEHICLE HAZARDS	
- Reactor radiation	Use proper shielding and prox operations procedures

Depot Advantages In Safety Hazards Abatement

The operational advantages for employing a propellant depot to reduce the effects of potential safety hazards on-orbit are listed as alternatives for Mars mission risk abatement.

Depot Advantages in Safety Hazards Abatement

ADVANTAGES

- LESS CHANCE OF LOSS BECAUSE DEPOT PROVIDES BETTER SPACE DEBRIS AND MICROMETEROID SHIELDING THAN VEHICLE
- DRY TANKS ON VEHICLE CAN BE REPLACED IF VEHICLE DEBRIS COLLISION
- LESS RMS CONTROL REQUIREMENTS FOR ASSEMBLY NODE BECAUSE OF VEHICLE DRY TANKS
- DEPOT CAN PROVIDE FOR NUCLEAR PROCESSING ON NTR RETURN FROM MISSION

SECTION 6

PROPELLANT DEPOT TECHNOLOGY REQUIREMENTS

There are a number of technology issues that must be resolved in order for a zero-g cryogenic propellant storage depot to be viable. Other technology issues warrant attention because they could provide cost and operational advantages if resolved. Most of the cryogenic fluid management issues presented here must be addressed with orbital flight experiments. A summary of the depot technology needs and their criticality ratings are presented in the charts "Depot Technology Needs".

There are five thermal control issues: degradation of insulating material due to exposure to space environment, development of low thermal conductivity materials, degradation in the performance of thick Multi-Layer Insulation (MLI) due to launch effects, development of para to ortho converters, and development of multiple/coupled Vapor Cooled Shields (VCS). Understanding the degradation of the insulation due to launch and long term space environment effects will prevent the need for overdesigning insulation systems to ensure adequate performance. Development of low thermal conductivity materials can reduce the heat leak into cryogenic tanks through penetration, such as supports, plumbing, and electrical lines. Para/ortho converters make use of the endothermic process that hydrogen undergoes in changing from from its para to ortho form. This process occurs by itself as para hydrogen vapor warms up. However, it is too slow to be of any use without the aid of a catalyst. By converting vented hydrogen in a para/ortho converter, heat that otherwise would have gone into the tank can be absorbed and then dumped overboard. Development of a multiple/coupled VCS would also provide for more efficient venting by intercepting incoming heat with the vent gas before it goes overboard. All of these thermal control issues have the payoff of reducing boil-off and/or insulation system weight.

The issues associated with pressure control are: predicting and increasing Thermodynamic Vent System (TVS) performance to allow efficient control of tank pressure by venting only vapor, determining the amount of fluid mixing required to control stratification/rapid pressure rise in the tank, and developing refrigeration or reliquifaction systems to convert the boil-off back to liquid. A TVS is a device for controlling tank pressure. Liquid from the tank flows through a Joule-Thompson valve where its pressure (and temperature) are reduced at constant enthalpy. This colder two phase fluid then flows through a heat exchanger in the tank to absorb heat, thereby reducing the tank pressure. The fluid exiting the TVS can then be supplied to a VCS, which is embedded in the MLI. Thus, a TVS and VCS combination provides a very efficient method of controlling tank pressure. If the fluid in the tanks stratifies, then venting can be delayed by mixing the tank contents to bring the liquid and ullage into thermodynamic equilibrium. The minimum amount of fluid mixing must be determined so that excess energy is not added to the tank contents. Refrigeration or reliquifaction systems are an alternative to venting, where the boil-off is eliminated or reliquified. The drawback of these systems are that they require electrical power and they are quite heavy. Also, reliable, long-life space qualified systems do not exist. The payoff from these pressure control issues is a large reduction in the amount of cryogens that would be vented/dumped trying to control tank pressure in zero-g.

The liquid acquisition issues are: predicting Liquid Acquisition Device (LAD) performance and determining the heat leak threshold where the screen dries out causing LAD failure. (Heat leak enters the LAD through its supports.) This technology must be developed in order to be able to reliably obtain single phase liquid from cryogenic depot tanks.

Pressurization issues include: determination of pressurant collapse in zero-g so that autogenous pressurization systems can be properly sized and development of long life space qualified cryogenic pumps and/or compressors (for both the autogenous pressurization system and transfer system). The amount of pressurant collapse in zero-g is not known and bounding calculations show an order of magnitude uncertainty in the amount of pressurant required with relatively high liquid fill levels. The payoffs in this area are weight and operational savings with properly sized systems.

Advanced instrumentation needs development in the areas of: quantity gauging to determine liquid inventories in zero-g, mass flow/quality metering to aid in determining the amount of propellant transferred, leak detection devices that operate in space, and liquid/vapor sensors for determining liquid/vapor interfaces in tanks and lines. Development of zero-g quantity gauges is essential for determining storage depot inventories, while the other instruments simplify operations and maintenance.

Liquid handling issues include: understanding how to control liquid dynamics/slosh due to docking and reboost perturbations to the zero-g environment and determination of fluid dumping/tank inerting procedures for emergencies. The latter being extremely important for safety considerations.

Liquid transfer technology issues are: development of efficient transfer line and tank chilldown methods, verification that acceptably high fill levels can be achieved with the no-vent fill process, and determination of how to successfully fill a LAD in zero-g. The payoff of efficient chilldown processes is propellant and possibly operational savings. The verification of no-vent fill and LAD fill procedures are essential for a zero-g cryogenic storage depot.

The development of efficient O2/H2 thrusters for reboost offers a sizable weight savings over other options (such as storable propellant thrusters) which have a lower specific impulse, Isp. These thrusters would also eliminate the need for resupply of large quantities of storable propellants.

SECTION 6

PROPELLANT DEPOT TECHNOLOGY REQUIREMENTS

Depot Technology Needs

Depot technology needs are presented with their issues and a criticality rating. Thermal Control issues include: degradation of insulating material due to exposure to space environment, development of low thermal conductivity materials in order to reduce the heat input to the cryogen tanks, degradation of thick Multi-Layer Insulation (MLI) due to launch effects, development of para to ortho converters to utilize the endothermic process of hydrogen changing from its para to ortho states, and development of multiple/coupled Vapor Cooled Shields (VCS) to reduce the heat leak into the cryogens. Resolution of these Thermal Control issues have the payoff of reducing boil-off and/or insulation system weight.

Pressure Control issues are: predicting/increasing Thermodynamic Vent System (TVS) performance to allow efficient control of tank pressure by venting only vapor, determining the amount of fluid mixing required to control stratification/rapid pressure rise in the tank, and developing refrigeration/reliquifaction systems to convert the boil-off back to liquid. The payoff from these issues is a large reduction in the amount of cryogens that would be vented/dumped trying to control tank pressure in zero-g.

Liquid Acquisition issues are: predicting Liquid Acquisition Device (LAD) performance and determining the heat leak threshold where the screen dries out causing LAD failure. This technology must be developed in order to be able to reliably obtain single phase liquid from the depot tanks.

Pressurization issues include: determination of pressurant collapse in zero-g so that autogenous pressurization systems can be properly sized and development of long life space qualified cryogenic pumps and/or compressors (for both the pressurization system and transfer system). The payoffs in this area are weight and operational savings with properly sized systems.

GENERAL DYNAMICS
Space Systems Division

Depot Technology Needs

Technology	Issues	Criticality*
Thermal Control	Degradation of Material	2
	Low Thermal Conductivity Materials	2
	Launch Effects on Thick MLI	2
	Para/Ortho Conversion	3
	Multiple/Coupled VCS	2
Pressure Control	TVS Performance	1
	Fluid Mixing for Stratification Control	1
	Refrigeration/Reliquifaction	2
Liquid Acquisition	LAD Performance	2
	LAD Performance with Heat Input	3
Pressurization	Autogenous Pressurization	2
	Mechanical Pumps/Compressors	2

* **Level 1 (Enabling):** Cannot be Configured Without this Technology/Safety Critical
Level 2 (High Impact): Significant System Improvement or Reduced Operational Complexity/Cost
Level 3 (Enhancing): Modest System Performance Improvement

Depot Technology Needs (Con't)

Depot technology needs are continued with their issues and a criticality rating. Instrumentation needs development in the areas of: quantity gauging to determine liquid inventories in zero-g, mass flow/quality metering to aid in determining the amount of propellant transferred, leak detection devices that operate in space, and liquid/vapor sensors for determining liquid/vapor interfaces in tanks and lines. Development of zero-g quantity gauges is essential, while the other instruments simplify operations.

Liquid Handling issues include: understanding how to control liquid dynamics/slosh due to docking and reboost perturbations to the zero-g environment and determination of fluid dumping/tank inerting procedures for emergencies. The latter being extremely important for safety considerations.

Liquid Transfer technology issues are: efficient transfer line and tank chilldown, verification that acceptably high fill levels can be achieved with the no-vent fill process, and determination of how to successfully fill a LAD. The payoff of the efficient chilldown processes is a savings of propellant. The no-vent fill and LAD fill are essential for a zero-g depot.

Development of efficient O_2/H_2 thrusters for reboost offer a sizable weight savings over other options which have a lower Isp.

Depot Technology Needs (Con't)

GENERAL DYNAMICS
Space Systems Division

Technology	Issues	Criticality*
Instrumentation	Quantity Gauging	1
	Mass Flow/Quality Metering	3
	Leak Detection	2
	Liquid/Vapor Sensors	3
Liquid Handling	Liquid Dynamics/Slosh Control	2
	Fluid Dumping and Tank Inerting	1
Liquid Transfer	Transfer Line Chilldown	3
	Tank Chilldown with Spray	3
	No-Vent Fill	1
	LAD Fill	1
O2/H2 Thrusters	Efficient Gas Thruster for Reboost	2

* **Level 1 (Enabling):** Cannot be Configured Without this Technology/Safety Critical
Level 2 (High Impact): Significant System Improvement or Reduced Operational Complexity/Cost
Level 3 (Enhancing): Modest System Performance Improvement

SECTION 7

ORBITAL PROPELLANT DEPOT COST ASSESSMENT

7.1 GROUNDRULES & ASSUMPTIONS

The cost section of this study is set up to evaluate the differences in Life Cycle Cost for various mission scenarios. Each case is set up to accommodate the Minimum Science of Full Science manifest schedule. A summary is given in the charts "Cost Groundrules and Assumptions". All costs are ROM estimates for preliminary planning and trade study comparison purposes only. The costs are presented in constant year 1990 millions of dollars. It is assumed that all of the technology needed for the tasks outlined is available. The costs associated with ground facilities are not included in any of the estimates. In addition, a probability of 100% launch success is assumed.

A range of Earth-To-Orbit (ETO) vehicle costs were given by Boeing Aerospace: $1,000/lb to $3000/lb. The $1,000/lb figure was used for this study. The exception to the above is the first launch associated with the depot. This launch only takes up the structures and mechanisms, without tanks or fuel, and is much lighter. A $150M launch vehicle cost allocation is used for this flight (in each appropriate mission scenario). The DDT&E costs include one shipset of a production article for design evaluation and test, and 0.75 of a shipset for initial spares. The logistics spares have not been included in the cost estimate. Other figures used in the estimate include; EVA rate at $201 K/hr, IVA rate at $23 K/hr, learning curve of 90%, and a rate curve of 95%.

7.2 METHODOLOGY

The cost estimating methodology incorporates a parametric cost model as the principal tool for development of life-cycle cost estimates, as shown in the chart "Cost Methodology". Parametric models can be used to efficiently produce credible cost estimates with limited input data and design definition typically available early in the study process. The process used consists of developing cost estimating relationships (CERs) for each WBS element using our cost and technical data base, gathering specific technical data, and entering the CERs and data into the cost model.

7.3 WORK BREAKDOWN STRUCTURE

The Work Breakdown Structure (WBS) shows the framework used for this study, as shown in the chart "Work Breakdown Structure". It includes the basic structures for each defined element, the associated production support (program management, systems engineering, tooling, integration & checkout, and ground support equipment), and required operations. Costs associated with other operational functions (mission operations, logistics, ground launch operations, etc.) are assumed to be included in the ETO vehicle costs.

7.4 COST SCENARIOS

The cost summaries are broken down by mission scenarios, as shown in the chart "Cost Scenarios Considered". The Cryo All-Propulsive (CAP) MTV is used for the Minimum Science Scenario and the Nuclear Thermal Rocket (NTR) MTV is used for the Full Science Scenario. Three options were considered for each scenario. The particular manifests for each option, and sensitivity, is discussed in the trade study analysis section.

7.5 LIFE CYCLE COST

The Life Cycle Costs (LCC) for the missions associated with the Minimum Science Scenario are shown on the "Life Cycle Cost Overall Summary - Minimum Science with CAP" chart. The LCC of the direct launch of full CAP fuel tanks is $8,732M, which is the most cost effective case. The top-off from tanker case only differs by the cost to develop and produce the tankers, whereas the top-off from depot case requires development and production of the depot as well as additional ETO launches.

The Life Cycle Costs (LCC) for the missions associated with the Full Science Scenario are shown on the "Life Cycle Cost Overall Summary - Full Science with NTR" chart. Again, the LCC of the direct launch of full NTR fuel tanks is the most cost effective case at $16,540M.

The chart titled "Life Cycle Cost Breakdown – Minimum Science with CAP" shows a funding profile for the top-off from depot case supporting the Minimum Science Scenario. The IOC for this case is 2012 with the first flight for depot delivery. The dip in the curve that occurs in 2018 is due to the amount of time between depot tank delivery and the spread based on our beta distribution family data base. The peak funding occurs in 2012 at $1,490M. A breakdown of the recurring portion of this chart is shown in the following chart, "Depot Recurring Cost Breakdown – Minimum Science with CAP". A similar LCC funding profile for the Full Science Scenario, tanker top-off case is given in the "Life Cycle Cost Breakdown – Full Science with NTR" chart. The IOC for this scenario is 2008, with the peak funding in 2018 at $1,360M.

7.6 COST SENSITIVITY TO MTV BOILOFF RATE

The boil-off sensitivity charts "Boiloff Sensitivities - Minimum Science (Depot)" and "Boiloff Sensitivities - Minimum Science (Tanker)" summarize the results of the CAP fuel tank boil-off rate sensitivities for the depot and tanker top-off cases, respectively. The LCC of the baseline cases, which are the same as seen on the "Life Cycle Cost Overall Summary - Minimum Science with CAP" chart, are $15,294M for the depot and $9,612 for the tanker, which are the most cost effective cases. The cost differences are mainly due to the difference in the total number of ETO launches, the number of fuel tanks necessary, and the number of years spread for the fuel tank flights.

In addition, the difference between the LCC for the depot top-off case and tanker top-off case becomes smaller as the MTV boil-off rate increases. The approximate intercept is at a MTV boiloff rate of 6.5%.

7.7 CONCLUSIONS AND RECOMMENDATIONS

The "Cost Conclusions and Recommendations" chart summarize the results. The direct launch of the Mars Transfer Vehicle tanks is the most cost effective case in both the Minimum Science Scenario and the Full Science Scenario. This is due to the fact that the direct launch case always requires the least number of Earth-to-Orbit Vehicle launches. In addition, there are some costs which have been excluded from the tanker and depot cases (i.e. the ground production facilities cost) which would increase the cost differences. There is a higher technology risk associated with the depot and tanker cases and a greater risk for failures due to the number of fuel transfers required. The benefits, however, might outweigh the risks. By using a depot there would be a decreased risk in mission completion and scheduled delay due to the storage of extra fuel for the MTV. Further system definitions and sensitivity studies are recommended to determine optimized depot and tanker scenarios.

SECTION 7

ORBITAL PROPELLANT DEPOT COST ASSESSMENT

Cost Groundrules and Assumptions

The cost section of this study is set up to evaluate the differences in Life Cycle Cost for various mission scenarios. Each case is set up to accommodate the Minimum Science of Full Science manifest schedule. All costs are ROM estimates for preliminary planning and trade study comparison purposes only. The costs are presented in constant year 1990 millions of dollars. It is assumed that all of the technology needed for the tasks outlined is available. The costs associated with ground facilities are not included in any of the estimates. In addition, a probability of 100% launch success is assumed.

Cost Groundrules and Assumptions

- Model Based In Millions Of CY 1990 Dollars
- ROM Estimates For Trade Study And Comparison Purposes Only
- Escalation Based On NASA R&D Programs Index Dated 3-31-90
- Costs Do Not Include Government Support Or Contractor Fee
  -- 30% Cost Contingency Was Added
- Excluded Ground Facility Costs (To Build/Store Depot & Tanker Parts)
- Assume Existing Ground Monitoring Station Used For Depot
- Excluded Cost For Getting Astronauts Up To Depot Assembly Site
- Assume All Of The Technology Needed Is Available
- No Cost Allowance For Launch Failures
- IOC For Each Scenario Assumed To Be Year Of First ETO Flight

Cost Groundrules and Assumptions (Con't)

A range of Earth-to-Orbit (ETO) vehicle costs were given by Boeing Aerospace: $1,000/lb to $3000/lb. The $1,000/lb figure was used for this study. The DDT&E costs include one shipset of a production article for design evaluation and test, and 0.75 of a shipset for initial spares. The logistics spares have not been included in the cost estimate. Other figures used in the estimate include; EVA rate at $201 K/hr, IVA rate at $23 K/hr, learning curve of 90%, and a rate curve of 95%.

Cost Groundrules and Assumptions (Con't)

- Earth-to-Orbit Vehicle Costs Provided By Boeing Aerospace:
  - $1,000/lb @ 120mT => $264.6 M Allocated To Each ETO Flight

    (First Depot Flights Received An ETO Vehicle Allocation Of $150M For A Smaller Launch Vehicle)
  - Assume Following Costs Included In Above Figure:
    - Mission/Launch Operations
    - Logistics
    - Ground Launch Operations
    - Integration - Checkout

- Included In DDT&E Costs:
  - 1 Design Evaluation Test Article
  - 0.75 Shipset Allocated For Initial Spares

- Logistics Spares Excluded

- Production Hardware Learning At 90%, Rate Curve = 95%

- Orbital Maintenance Hourly Rates:
  - EVA = $201 K/Hr
  - IVA = $23 K/Hr

- All Resupply Flights For LTV Tanks Included In NTR Flight Scenario

Cost Methodology

The cost estimating methodology incorporates a parametric cost model as the principal tool for development of life-cycle cost estimates. Parametric models can be used to efficiently produce credible cost estimates with limited input data and design definition typically available early in the study process. The process used consists of developing cost estimating relationships (CERs) for each WBS element using our cost and technical data base, gathering specific technical data, and entering the CERs and data into the cost model.

Cost Methodology

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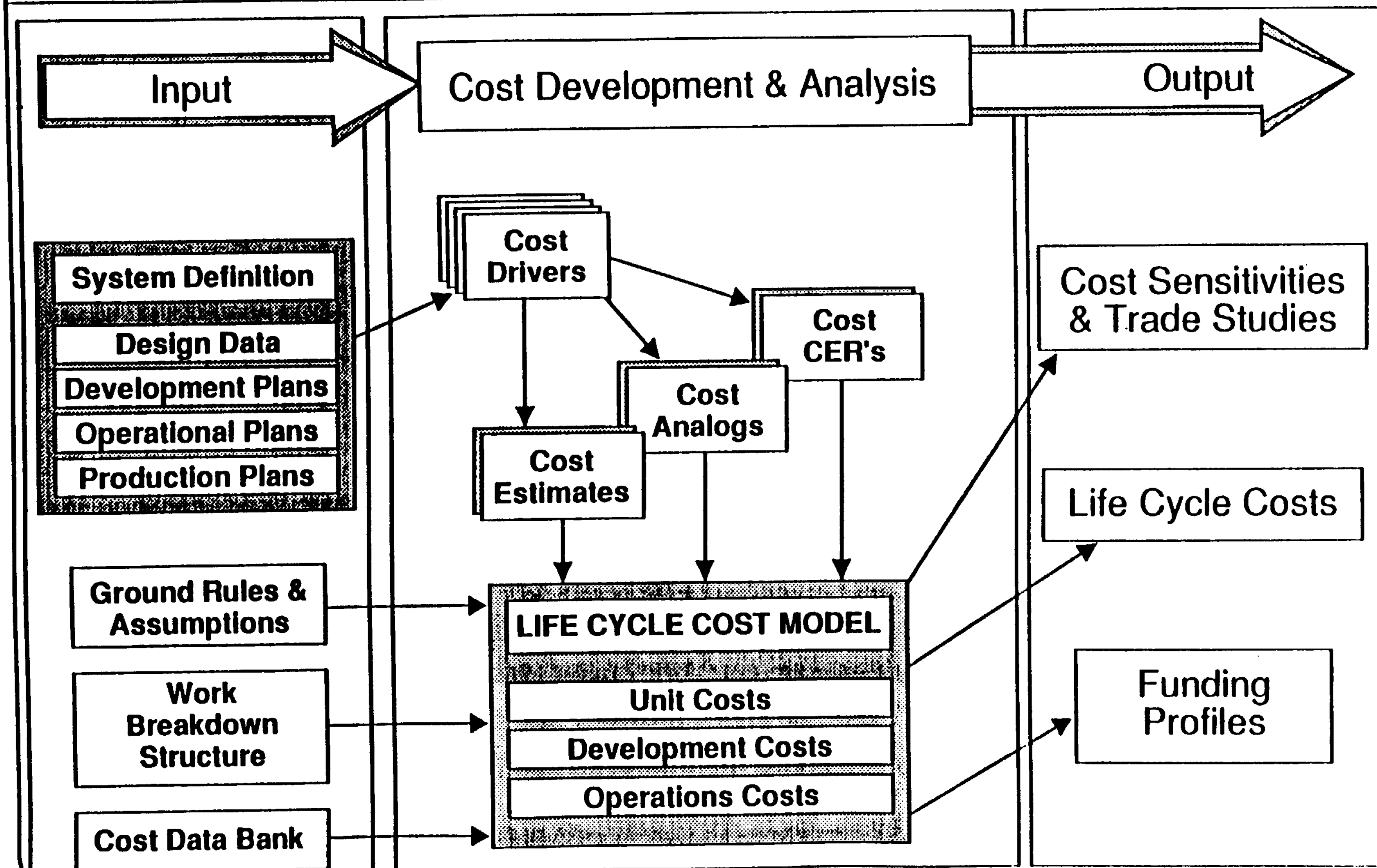

VG7-03

Work Breakdown Structure

The Work Breakdown Structure (WBS) shows the framework used for this study. It includes the basic structures for each defined element, the associated production support (program management, systems engineering, tooling, integration & checkout, and ground support equipment), and required operations. Costs associated with other operational functions (mission operations, logistics, ground launch operations, etc.) are assumed to be included in the ETO vehicle costs.

Work Breakdown Structure

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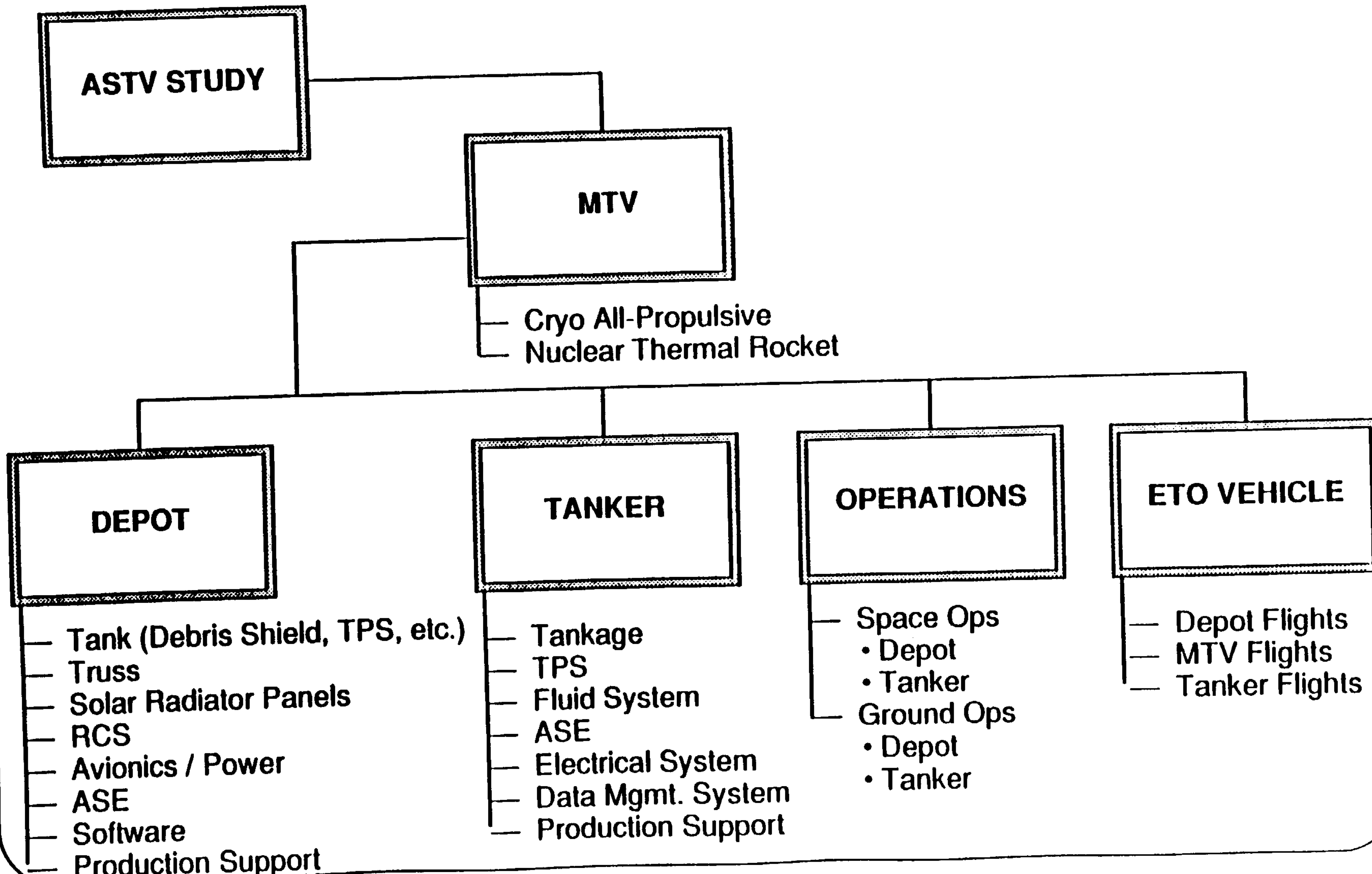

VG7-04

Cost Scenarios Considered

The cost summaries are broken down by mission scenarios. The Cryo All-Propulsive (CAP) MTV is used for the Minimum Science Scenario and the Nuclear Thermal Rocket (NTR) MTV is used for the Full Science Scenario. Three options were considered for each scenario. The particular manifests for each option, and sensitivity, is discussed in the trade study analysis section.

Cost Scenarios Considered

Minimum Science Scenario

Mars Transfer Vehicle: Cryo All-Propulsive (CAP)

- **Top-Off From Depot**
- **Top-Off From Tanker**
- **Direct Launch of Fuel**

Plus Additional Boil-Off Rate Sensitivities To These Cases

Full Science Scenario

Mars Transfer Vehicle: Nuclear Thermal Rocket (NTR)

- **Top-Off From Depot**
- **Top-Off From Tanker**
- **Direct Launch of Fuel**

Life Cycle Cost Overall Summary - Minimum Science with CAP

This chart summarizes the Life Cycle Costs (LCC) for the missions associated with the Minimum Science Scenario. The LCC of the direct launch of full CAP fuel tanks is $8,732M which is the most cost effective case. The top-off from tanker case only differs by the cost to develop and produce the tankers, whereas the top-off from depot case requires development and production of the depot as well as additional ETO launches.

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Life Cycle Cost Overall Summary - Minimum Science with CAP

SCENARIO: Minimum Science

	LCC		
	Direct Launch of Fuel	Top-off From Depot	Top-off From Tanker
Depot			
Structures	—	$ 3,363 M	—
Production Support	—	2,428 M	—
Operations	—	92 M	—
Tanker			
Structures	—	—	$ 539 M
Production Support	—	—	350 M
Operations	—	—	30 M
ETO			
Depot Flights	—	$ 3,325 M	—
CAP Flights	$ 8,732 M	6,086 M	$ 5,557 M
Tanker Flights	—	—	3,175 M
TOTAL —>	$ 8,732 M	$ 15,294 M	$ 9,612 M

Life Cycle Cost Overall Summary - Full Science with NTR

This chart summarizes the Life Cycle Costs (LCC) for the missions associated with the Full Science Scenario. The LCC of the direct launch of full NTR fuel tanks is $16,540M which is the most cost effective case. The top-off from tanker case only differs by the cost to develop and produce the tankers, whereas the top-off from depot case requires development and production of the depot as well as additional ETO launches.

Life Cycle Cost Overall Summary - Full Science with NTR

SCENARIO: <u>Full Science</u>

	LCC		
	Direct Launch of Fuel	Top-off From Depot	Top-off From Tanker
Depot			
Structures	—	$ 5,478 M	—
Production Support	—	2,477 M	—
Operations	—	149 M	—
Tanker			
Structures	$ 366 M	—	$ 582 M
Production Support	268 M	—	371 M
Operations	30 M	—	52 M
ETO			
Depot Flights	—	$ 4,912 M	—
NTR Flights	$ 14,818 M	11,642 M	$ 12,569 M
Tanker Flights	1,058 M	—	3,572 M
TOTAL —>	$ 16,540 M	$ 24,658 M	$ 17,146 M

Life Cycle Cost Breakdown - Minimum Science with CAP

This is a funding profile chart for the top-off from depot case supporting the Minimum Science Scenario. The IOC for this case is 2012 with the first flight for depot delivery. The dip in the curve that occurs in 2018 is due to the amount of time between depot tank delivery and the spread based on our beta distribution family data base. The peak funding occurs in 2012 at $1,490M.

Life Cycle Cost Breakdown - Minimum Science with CAP

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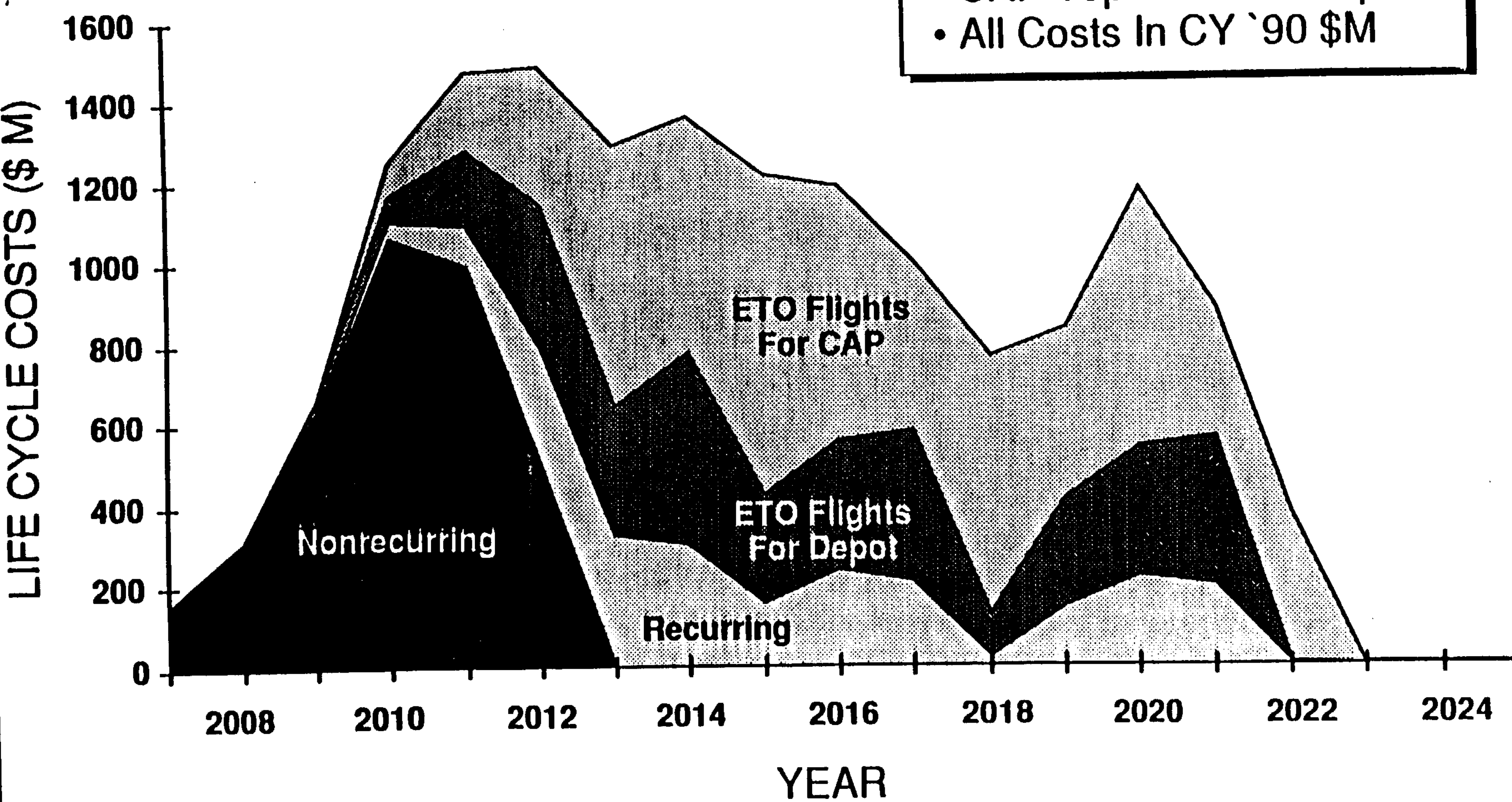
- CAP Top-off From Depot
- All Costs In CY `90 $M



VG7-08

Depot Recurring Cost Breakdown - Minimum Science with CAP

This funding profile chart shows a breakdown of the recurring section of the previous LCC chart. The number of ETO flights is the major cost driver and the production of depot tanks is second. The dip in the curve that occurs in 2018 is due to the amount of time between depot tank delivery and the spread based on our beta distribution family data base. The peak funding occurs in 2014 at $775M.

Depot Recurring Cost Breakdown - Minimum Science with CAP

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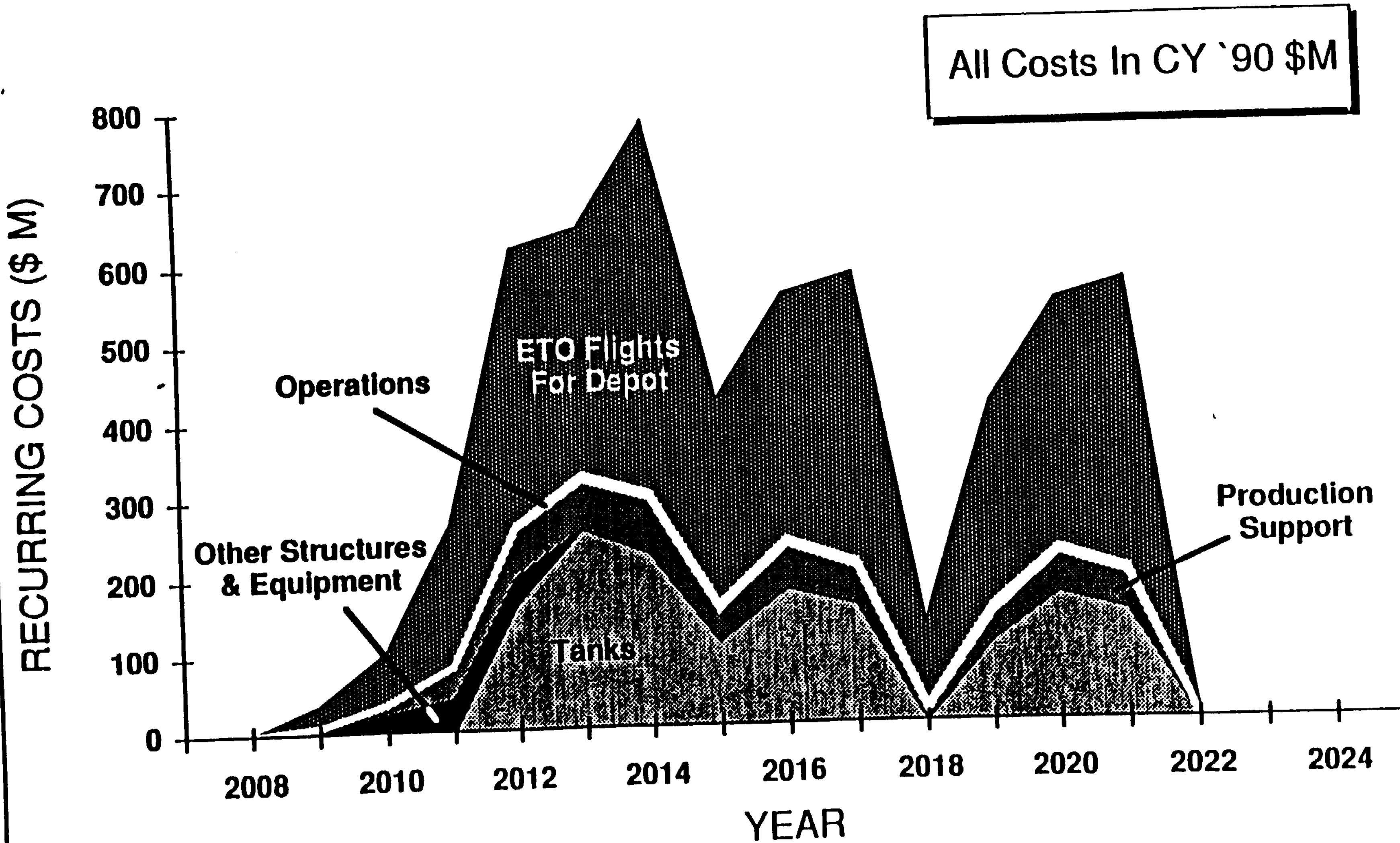


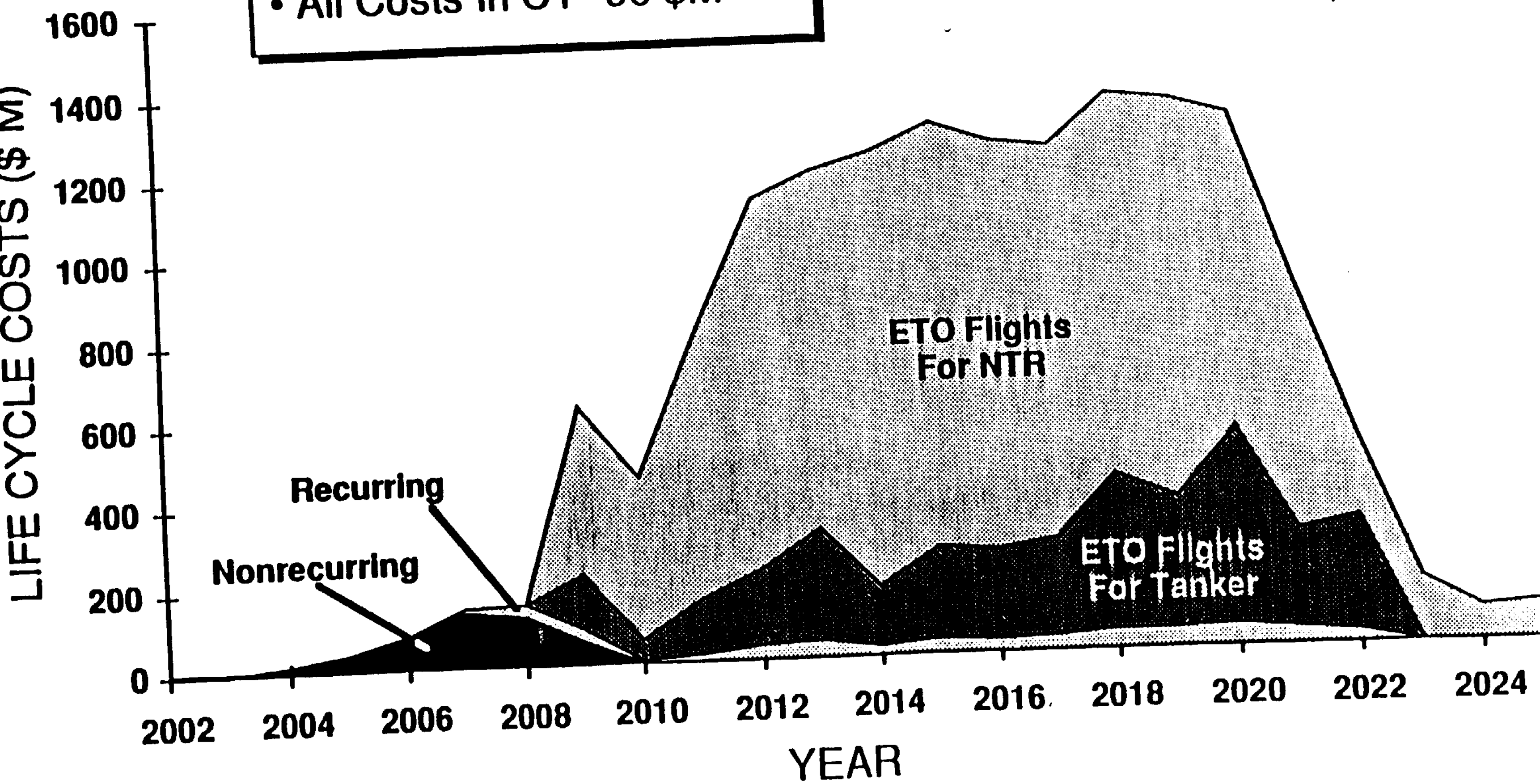
VG7-09

Life Cycle Cost Breakdown - Full Science with NTR

This is a funding profile chart for the top-off from tanker case supporting the Full Science Scenario. IOC for this case is 2008 with the first flight for the NTR vehicle. The peak funding occurs in 2018 at $1,360M.

Life Cycle Cost Breakdown - Full Science with NTR

- NTR Top-off From Tanker
- All Costs In CY `90 $M



Boiloff Sensitivities - Minimum Science (Depot)

This chart summarizes the results of the CAP fuel tank boil-off rate sensitivities to the Depot Top-off case. The baseline case, which is the same as seen on the *"Life Cycle Cost Overall Summary - Minimum Science With CAP"* chart, is $15,294M, which is the most cost effective case. The cost differences are mainly due to the difference in the total number of ETO launches, the number of Depot tanks necessary, and the number of years spread for the Depot tank flights.

Boiloff Sensitivities
- Minimum Science (Depot)

SCENARIO:
Minimum Science
Top-off From Depot

	LCC		
	Baseline	Sensitivities	
	CAP B/O = 1.1%	CAP B/O = 2.2%	CAP B/O = 3.3%
Depot			
Structures	$ 3,363 M	$ 3,610 M	$ 3,873 M
Production Support	2,428 M	2,515 M	2,614 M
Operations	92 M	90 M	95 M
ETO			
Depot Flights	$ 3,325 M	$ 3,854 M	$ 4,648 M
CAP Flights	6,086 M	5,557 M	5,557 M
Tanker Flights	—	—	—
TOTAL —>	$ 15,294 M	$ 15,626 M	$ 16,787 M

Boiloff Sensitivities - Minimum Science (Tanker)

This chart summarizes the results of the CAP fuel tank boil-off rate sensitivities to the Tanker Top-off case. The baseline case, which is the same as seen on the *"Life Cycle Cost Overall Summary - Minimum Science With CAP"* chart, is $9,612, which is the most cost effective case. The cost differences are mainly due to the difference in the total number of ETO launches, the number of Tankers necessary, and the number of years spread for the Tanker flights.

Boiloff Sensitivities - Minimum Science (Tanker)

SCENARIO:
Minimum Science
Top-off From Tanker

	LCC		
	Baseline	Sensitivities	
	CAP B/O = 1.1%	CAP B/O = 2.2%	CAP B/O = 3.3%
Tanker			
Structures	$ 539 M	$ 587 M	$ 780 M
Production Support	350 M	373 M	465 M
Operations	30 M	34 M	41 M
ETO			
Depot Flights	—	—	—
CAP Flights	$ 5,557 M	$ 6,615 M	$ 5,557 M
Tanker Flights	3,175 M	3,704 M	6,350 M
TOTAL —>	$ 9,612 M	$ 11,313 M	$ 13,193 M

Cost Conclusions and Recommendations

The direct launch of the Mars Transfer Vehicle tanks is the most cost effective case in both the Minimum Science Scenario and the Full Science Scenario. This is due to the fact that the direct launch case requires the least number of Earth-to-Orbit Vehicle launches. There may be a higher technology risk associated with the depot and tanker cases and a greater risk for failures due to the number of fuel transfers required. The benefits, however, might outweigh the risks. By using a depot there would be a decreased risk in mission completion and scheduled delay due to the storage of extra fuel for the MTV. Further system definitions and sensitivity studies are recommended to determine optomized depot and tanker scenarios.

Cost Conclusions and Recommendations

CONCLUSIONS

- DIRECT LAUNCH OF MTV TANKS HAS THE LEAST COST FOR MINIMUM AND FULL SCIENCE SCENARIOS
  - BASED ON SUCCESS ORIENTED ANALYSIS
  - BENEFIT MARGIN NARROWS WITH RESPECT TO DEPOT FOR FULL SCIENCE MISSION
  - DEPOT BREAK EVEN POINT APPROXIMATELY 6.5% PER MONTH BOILOFF RATE FOR MARS VEHICLE
- MAJOR COST DRIVER IS ETO VEHICLE LAUNCH COST

RECOMMENDATIONS

- FURTHER COST SENSITIVITY STUDIES NEEDED: DEPOT TANK BOILOFF - TANKER BOILOFF - TANK SIZES - ETC
- MORE INDEPTH STUDY OF DEPOT AND TANKER APPLICATIONS AND DEFINITIONS

SECTION 8

CONCLUSIONS AND RECOMMENDATIONS

8.1 CONCLUSIONS

The bottom line cost analysis results of this study show the direct Earth to Orbit launch of the Mars Transfer Vehicle tanks full of propellants, adequate to accomplish the Mars mission without topping-off with additional propellants prior to mission departure, as being the most cost effective option for both the Minimum and Full Science Scenarios. This is because the direct launch case requires the least number of Earth-to-Orbit Vehicle launches in a success oriented approach to the operations and cost analyses. However, there may be higher on-orbit risks involved in the direct delivery and top-off delivery options because of potential vehicle loss due to possible puncture of the vehicle tankage by meteoroid or space debris impacts. Consideration of this risk may provide a different outcome in favor of a propellant depot to support the Mars missions. It was also shown that there is only a narrow margin of benefit in the direct launch and top-off options over the propellant depot for the full science scenario. And as the Mars vehicle boiloff rates approach 6.5% per month the propellant depot costs reach a break even point with the other propellant delivery options.

Other conclusions reached in this study related to depot applications are as follows:

- The depot should be located in low Earth orbit
- The depot should be separate from Space Station Freedom and operated in a co-orbiting mode
- The propellant transfer methods should include zero-g transfer capability

It was apparent that two launch pads will be needed to support the full science missions because less than 90 day launch centers are required. Possible surge capabilities due to catastrophic failures on-orbit also add to the need for at least two launch pads.

These conclusions are summarized in the chart "Orbital Propellant Depot Study Conclusions".

8.2 RECOMMENDATIONS

Further system definitions and sensitivity studies are recommended to determine optimized LEO depot and tanker scenarios. These studies should include the non-success oriented aspects of mission operations and possible outcomes.

Mars depot requirements need to be further examined and evaluated to define appropriate architectures for implementation of the Mars exploration program. These Mars depot definitions should be exploited to reveal commonalities impacting low Earth orbit facilities.

These recommendations are summarized in the chart "Orbital Propellant Depot Study Recommendations".

SECTION 8

CONCLUSIONS AND RECOMMENDATIONS

Orbital Propellant Depot Study Conclusions

The trade study evaluations concluded that the depot should be located in low Earth orbit, separate from Space Station Freedom and operated in a co-orbiting mode, and the propellant transfer methods should include zero-g transfer capability.

It was apparent that two launch pads will be needed to support the full science missions because less than 90 day launch centers are required. Possible surge capabilities due to catastrophic failures on-orbit also add to the need for at least two launch pads. It was also shown that there is only a narrow margin of benefit in the direct launch and top-off options over the propellant depot for the full science scenario. And as the Mars vehicle boiloff rates approach 6.5% per month the propellant depot costs reach a break even point with the other propellant delivery options.

Four LEO depot concepts were developed for the minimum and full science scenarios for the Cryo/AB and NTR vehicles. Mars depot scenarios to support a large, long-term mission were identified and discussed.

A Cryogenic Fluid Management flight experiment would be required to develop the technologies needed to support an on-orbit depot and MTV.

The bottom line cost analysis results of this study show the direct Earth to Orbit launch of the Mars Transfer Vehicle tanks full of propellants, adequate to accomplish the Mars mission without topping-off with additional propellants prior to mission departure, as being the most cost effective option for both the Minimum and Full Science Scenarios. This is because the direct launch case requires the least number of Earth-to-Orbit Vehicle launches in a success oriented approach to the operations and cost analyses. However, there may be higher on-orbit risks involved in the direct delivery and top-off delivery options because of potential vehicle loss due to possible puncture of the vehicle tankage by meteoroid or space debris impacts. Consideration of this risk may provide a different outcome in favor of a propellant depot to support the Mars missions.

Orbital Propellant Depot Study Conclusions

- Trade Study Conclusions
  - Depot Located in LEO
  - Depot Separate Free Flyer Co-Orbiting with SSF
  - Zero-g Transfer Method
- Mission Manifesting Evaluation
  - Two Launch Pads Required to Support Full-Science Mission
  - Depot Justification Depends on MTV Boiloff Rate
- Depot Concepts
  - Four LEO Depot Concepts Sized for CAP and NTR Vehicles
  - Mars Depot Supports a Large, Long-Term Mars Mission
  - On-Orbit Safety Hazards May Justify On-Orbit Depot
- Flight Experiment Required for Depot Technology Development
- Preliminary Evaluation Indicates Direct Launch of MTV Tanks is Cost Effective for Minimum and Full Science Scenarios

Orbital Propellant Depot Study Recommendations

Further system definitions and sensitivity studies are recommended to determine optimized LEO depot and tanker scenarios. These studies should include the non-success oriented aspects of mission operations and possible outcomes.

Mars depot requirements need to be further examined and evaluated to define appropriate architectures for implementation of the Mars exploration program. These Mars depot definitions should be exploited to reveal commonalities impacting low Earth orbit facilities.

Orbital Propellant Depot Study Recommendations

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- Further Systems Definition of LEO Depot Configurations
- Sensitivity Study to Optimize Configuration
- Evaluation of Mission Operation Variations
- Further Examine and Develop Architechtures for Mars Depots

for liquid oxygen storage for the chemical vehicles. Cryogenic storage issues relating to ECLSS fluids and lander/ascent vehicle propellants will remain as well. A preliminary technology schedule is presented for cryogenic fluid system development for Mars mission applications. The cryogenic fluid systems schedule includes Earth-based thermal control and selected component fluid management (tank pressure control, liquid acquisition device effectiveness, etc.) tests, as well as planned flight experiments to carry out system and subsystem development (selected components) and verification/validation tests. Many of the technology issues will be answered during the technology/advanced development work to be carried out for a Lunar program. The major technology obstacles to be overcome by an NEP storage system are in the areas of high reliability long term thermal control systems (particularily for the lander/ascent tanks), and orbital/flight operations (fluid transfer, acquisition, etc.).

Summary

As noted before, some of the identified critical and high leverage technology development issues are common across all of the major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H_2, and possibly O_2 for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique SEP technology issues center around efficient solar power systems and electric thruster/PPU development. Common enhancing technologies include cryogenic refrigeration (lander tanks), O_2-H_2 RCS, advanced in-space assembly techniques, higher Isp cryogenic engines, and advanced structural materials development.

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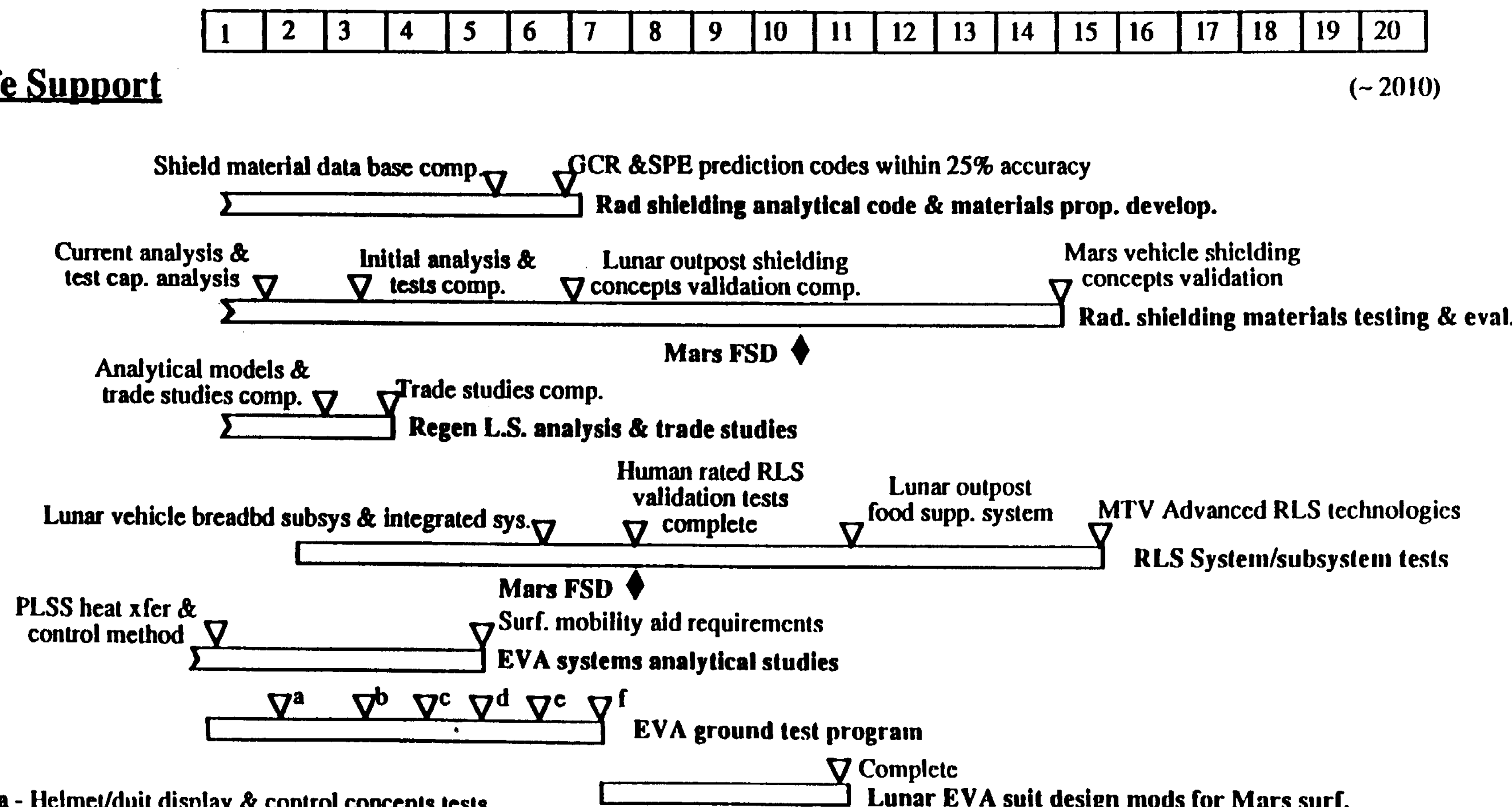
ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules

BOEING

Life Support



a - Helmet/duit display & control concepts tests
b - Lunar surf suit breadbd test
c - Gloves & displays in simul. SSF environment
d - Regen. PLSS breadbd for lunar surf.
e - Verif tests of adv. dexterious gloves & disp.
f - Complete breadbd lunar EVA suit / simulated surf. cond.

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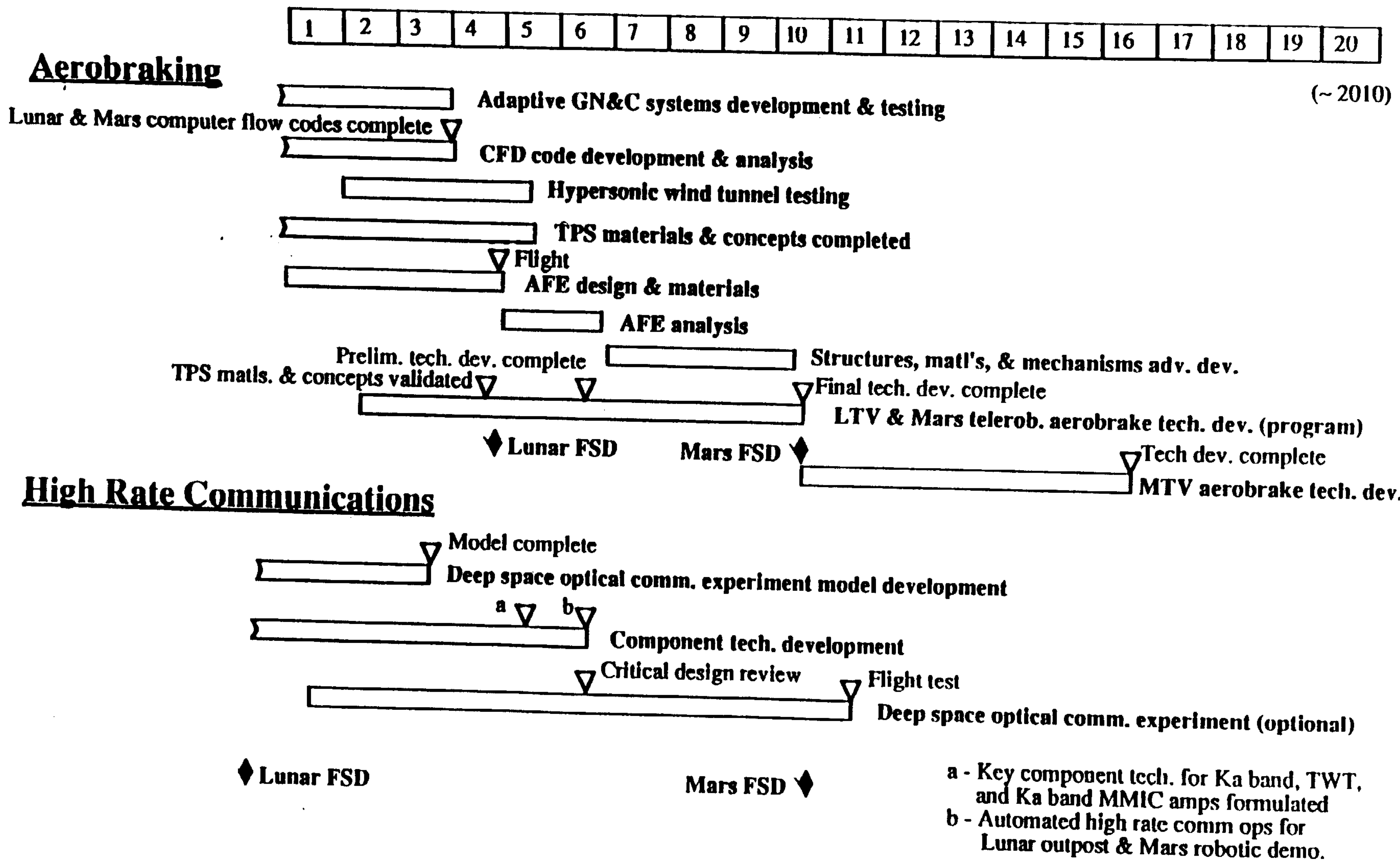
Preliminary SEI Technology Development Schedules (Cont.)



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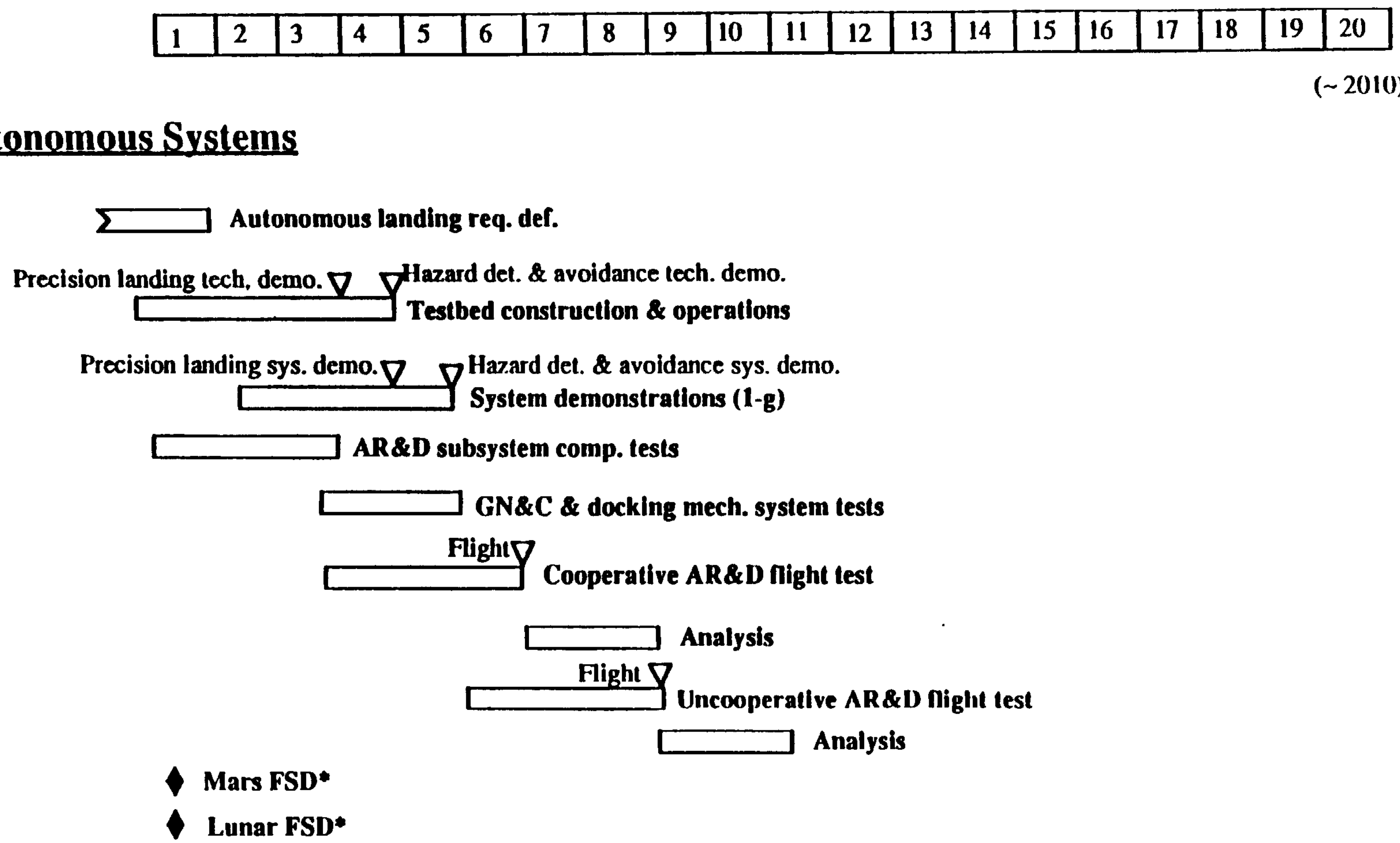
ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

BOEING



* Technology should not present FSD threatening problems; current technologies adequate for minimum mission.

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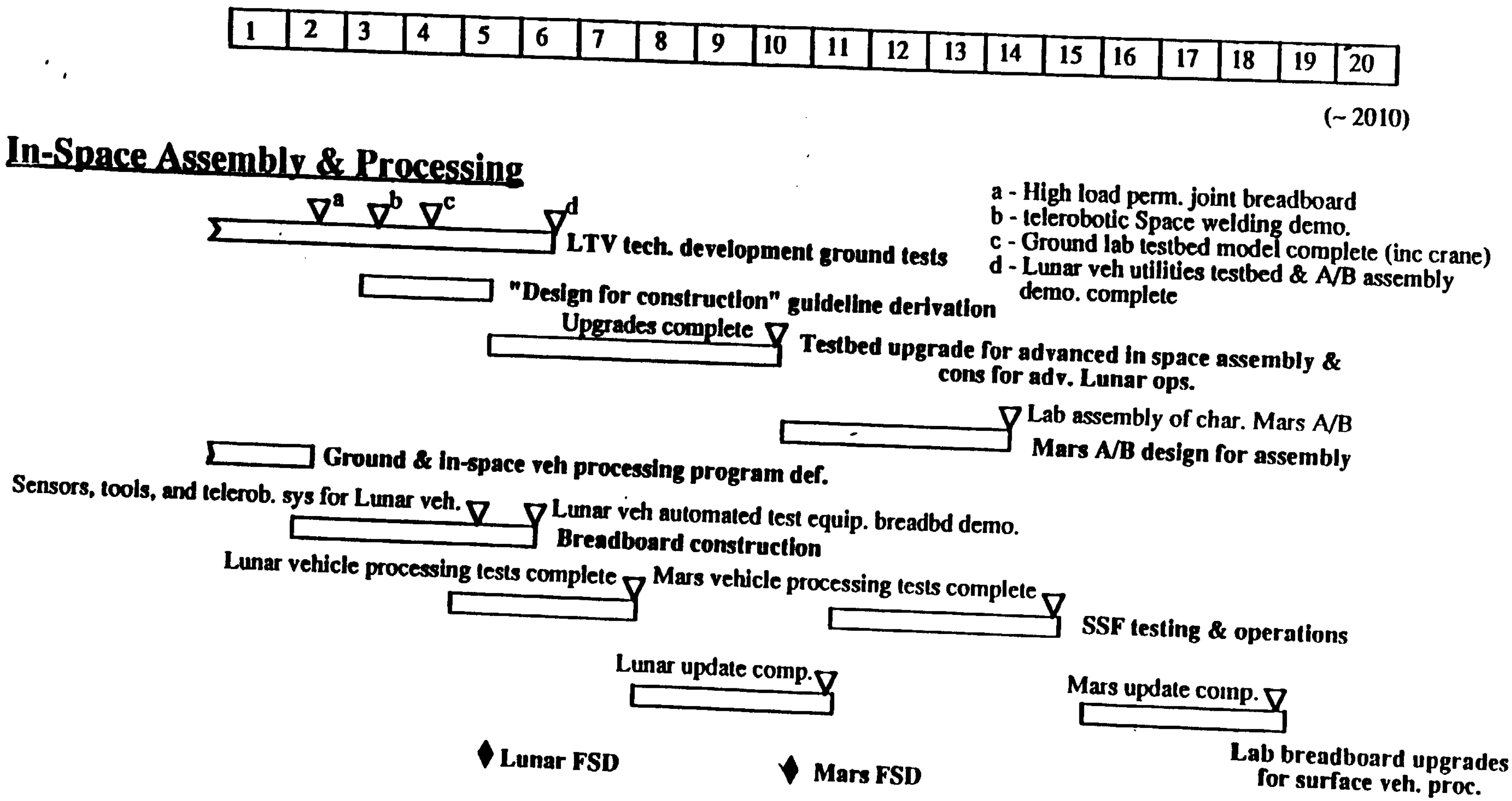
ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

BOEING



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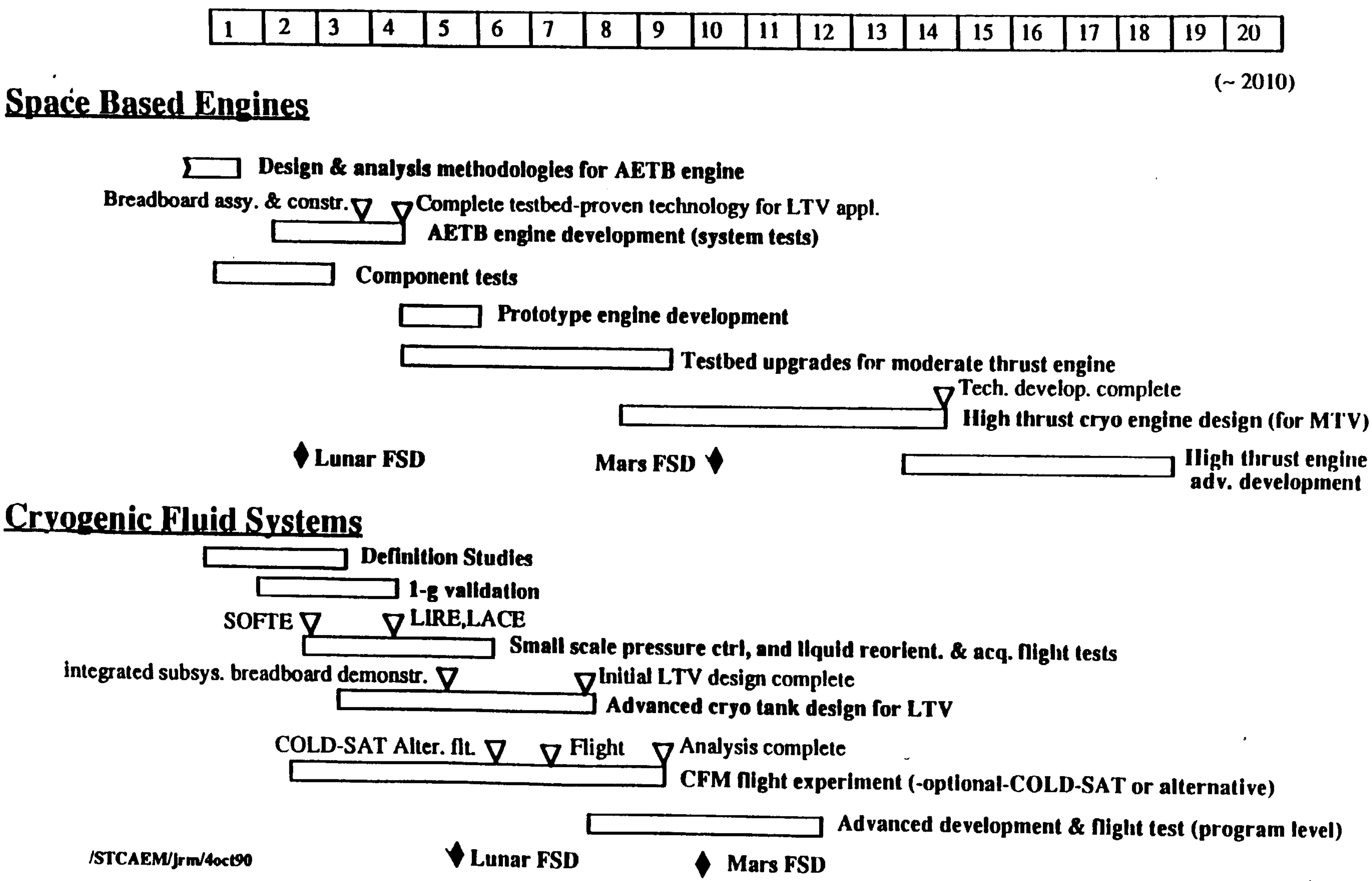
Preliminary SEI Technology Development Schedules (Cont.)



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Technology Development Concerns and Schedules - Nuclear Electric Propulsion (NEP)

Critical technology development issues relating to the reference NEP vehicle are presented in this section. Where applicable, the same charts are also included in the CAB, CAP, NTR, and SEP IP&ED documents. The focus of this section will be to bring out the most important issues relating to the reference NEP vehicle, and to present preliminary technology development schedules for these issues. The issues are presented here in outline form, beginning with the most important, with accompanying schedules wherever possible.

Nuclear Power System and Shielding Technology Development

One of the two most important areas of technology and advanced development for this vehicle option is the development of an integrated nuclear electric power system. A preliminary schedule for the development of a NEP propulsion system for a Mars vehicle is presented, which includes an integrated timeline for both of these technology development concerns. The schedule highlights both the point where a full scale development decision can be made (year 6), and when the first flight article will be available to the vehicle program (year 17). The most important area of development for the NEP option is the design, integration, and life testing of a space qualified multi-megawatt nuclear power system, capable of a 10 year lifetime. Major challenges to be overcome in the achievement of a long life efficient system lie in high temperature materials, liquid metal power conversion system development, and reactor design. In order to increase the efficiency of the power system, higher system temperatures are required. Materials capable of continuous operation above 1600K will be needed inside the reactor, and above 1500K in the conversion system components. Reactor design studies will focus on such technology concerns as high temperature fuel development, reactor and fuel designs with high burnup capability, high reliability control systems, and safing issues for flight operations. Long term life testing must be carried out for the power system (including reactor), to verify long term system reliability. A related technology development challenge for the program will probably be test facility design and development. Past space program nuclear tests were carried out in a testbed facility open to the atmosphere. Future test facilities must be closed in order to contain any fission products escaping from the system, as well as contain any perceived accident. This facility may prove to be very costly to build and operate. Nuclear electric propulsion offers a potential performance superior to the chemical and NTR vehicles, at the expense of a more costly and lengthy technology and advanced development program.

Electric Propulsion PPU/Thruster Technology Development

The second major area of technology development for the NEP is in large scale electric power processing unit (PPU), and thruster design and development. The development of long life PPU/thruster systems on a larger scale than currently available (MW level thrusters needed) is the major area of concern relating to the NEP concept. Thruster lifetimes on the order of a year or more (continuous) will be required for thrusters on the MW level in scale. Test facilities must be developed which are capable of supporting the long term life tests for these high power level thrusters. Finally, high temperature power processing equipment must be developed to increase system efficiency and reliability.

Life Support

A reliable, redundant long term life support system will be enabling for future exploration missions. The degree of closure of, and the reliability of the system are the

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major technology development concerns. Low-g human factors determination will also be an important technology consideration which will drive vehicle design. An integrated schedule of the major areas of the life support technology development task are presented. It includes radiation shielding and materials, regenerative life support, and EVA systems development. As before, the points where Lunar and Mars full scale development decisions can logically be made in the technology program are highlighted.

Aerobraking (low energy)

Low energy aerobraking will offer mission benefits in the areas of decreased demands on the descent propulsion system, and improved crossrange capability. This area presents a variety of issues for technology development including high strength to mass ratio structural materials, high temperature thermal protection systems (although not as high as for high energy aerobraking), avionics, assembly and operations, hypersonic test facilities and computer codes, and Mars atmosphere prediction. High strength structural material options include metal matrix composite, organic matrix composite, and advanced carbon-carbon elements. Other structural considerations include load distribution and attachment of payload for aerocapture, and ETO launch and assembly of large structures. Thermal protection systems issues include low mass ablative and reradiating materials, and structure/TPS integration issues. The aerobrake maneuver will place considerable demands on the vehicle avionics system with the need for real time trajectory analysis, and vehicle guidance and control. The launch and assembly of the large aerobrake structure will present ground and space assembly and ops problems which will require technology and advanced development in both the areas of design and operations. Finally, computational analysis and atmosphere prediction capability will be critical in the development of a man-rated aerobrake for Mars use. A preliminary development schedule for Lunar and Mars aerobrake technology development is presented. It includes the major milestones for both ground and flight testing. The points where a Lunar and Mars full scale development decision can be made are also highlighted on the schedule. It should be noted that this schedule was built with high energy aerobraking in mind, and will possibly be compressed to some degree if only low energy aerobraking is developed.

Vehicle Avionics and Software

Although the technology readiness level of vehicle avionics and software is ahead of many of the other technology areas listed in some respects, the demands on the system in the areas of processing rate, accuracy, autonomous operation, and status/health monitoring will drive technology and advanced development in areas not fully defined at this point. Software requirements cannot be fully determined until the vehicle design is at a more finished stage than the current levels. A preliminary schedule for autonomous systems development is presented. The decision points for full scale development The communications system options can be more fully defined before a final vehicle design is produced, however. A technology development schedule for advanced communications is presented. The NEP vehicle may not place the same level of demand on the avionics system in the area of trajectory analysis, but will likely place more demands on the system in the areas of status/health monitoring, and autonomous operation, fault diagnosis, and correction.

In-Space Assembly and Processing

The in-space assembly and processing of large space transfer vehicles will present a variety of technology advanced development challenges, particularly for the large LTV and MEV aerobrakes, and NEP vehicle. The large radiator structure, along with the many liquid metal pipe high pressure joints which must be made in orbit will present a variety of challenges in technology development (e.g. in-space welding), and assembly operations (e.g. robotics). As shown on the accompanying schedule, extensive ground tests must occur before any orbital work can be initiated. The vehicle designs will be driven to a large degree by the assembly facilities and technologies seen as being available during the vehicle buildup sequence. It should be noted that the schedule was not developed specifically for an NEP vehicle. Advances derived from this development process along with flight experience in earlier missions leading up to this evolutionary scenario could possibly accelerate the development plan considerably.

Cryogenic Fluid Management

The level of concern for technology development in the areas of cryogenic fluid management and storage will not be as for electric propulsion vehicles as for the high thrust systems, although many of the areas still remain important for the NEP vehicle. The Argon (or Zenon) propellant utilized for the electric propulsion system will be in a cryogenic liquid state, and will require long term storage and management technology levels similar to those for liquid oxygen storage for the chemical vehicles. Cryogenic storage issues relating to ECLSS fluids and lander/ascent vehicle propellants will remain as well. A preliminary technology schedule is presented for cryogenic fluid system development for Mars mission applications. The cryogenic fluid systems schedule includes Earth-based thermal control and selected component fluid management (tank pressure control, liquid acquisition device effectiveness, etc.) tests, as well as planned flight experiments to carry out system and subsystem development (selected components) and verification/validation tests. Many of the technology issues will be answered during the technology/advanced development work to be carried out for a Lunar program. The major technology obstacles to be overcome by an NEP storage system are in the areas of high reliability long term thermal control systems (particularily for the lander/ascent tanks), and orbital/flight operations (fluid transfer, acquisition, etc.).

Summary

As noted before, some of the identified critical and high leverage technology development issues are common across all of the major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H_2, and possibly O_2 for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique NEP technology issues center around nuclear power systems and electric thruster/PPU development. Common enhancing technologies include cryogenic refrigeration (lander tanks), O_2-H_2 RCS, advanced in-space assembly techniques, higher Isp cryogenic engines, and advanced structural materials development.

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ADVANCED CIVIL SPACE SYSTEMS

NEP Technology Development Program

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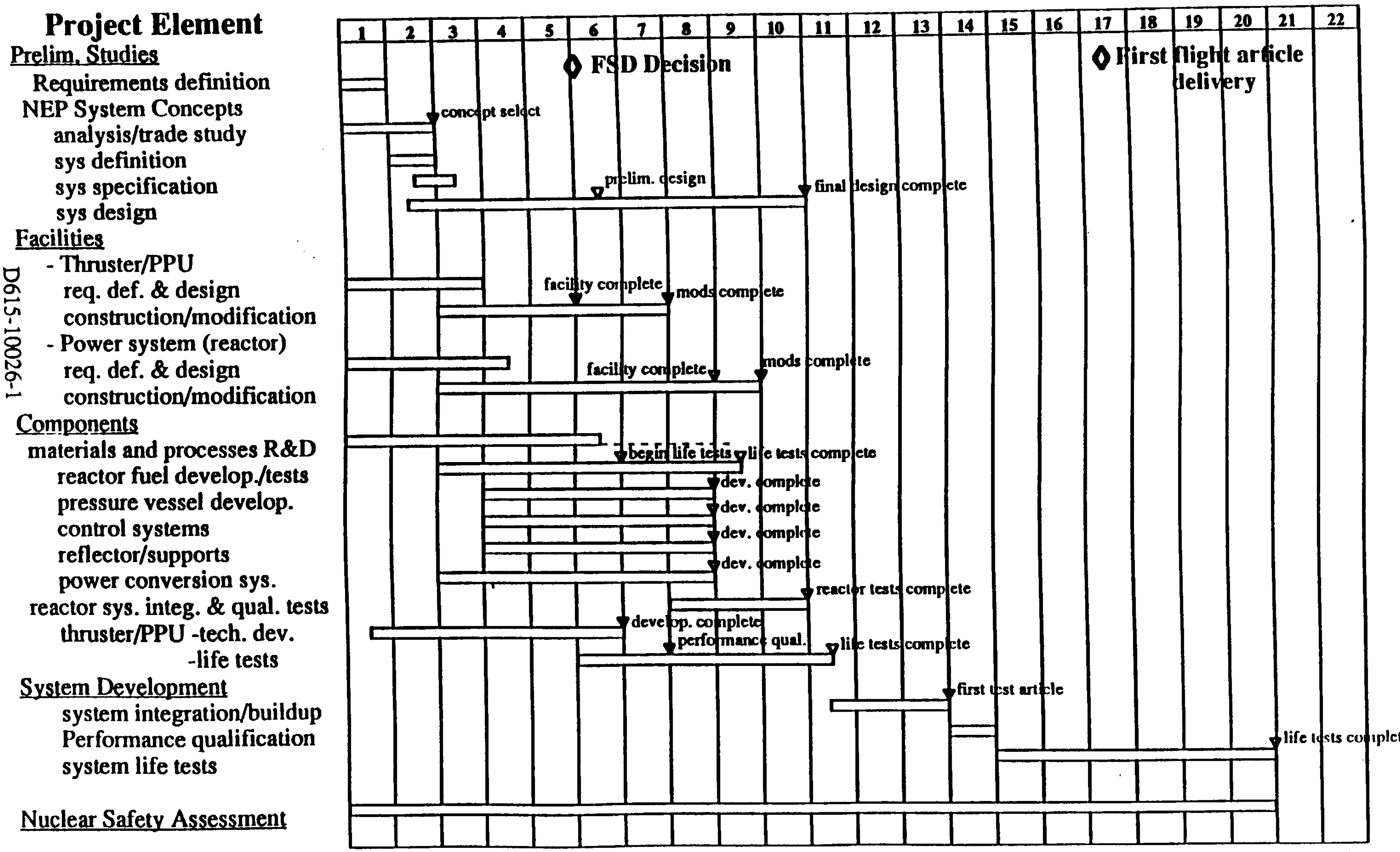

ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules

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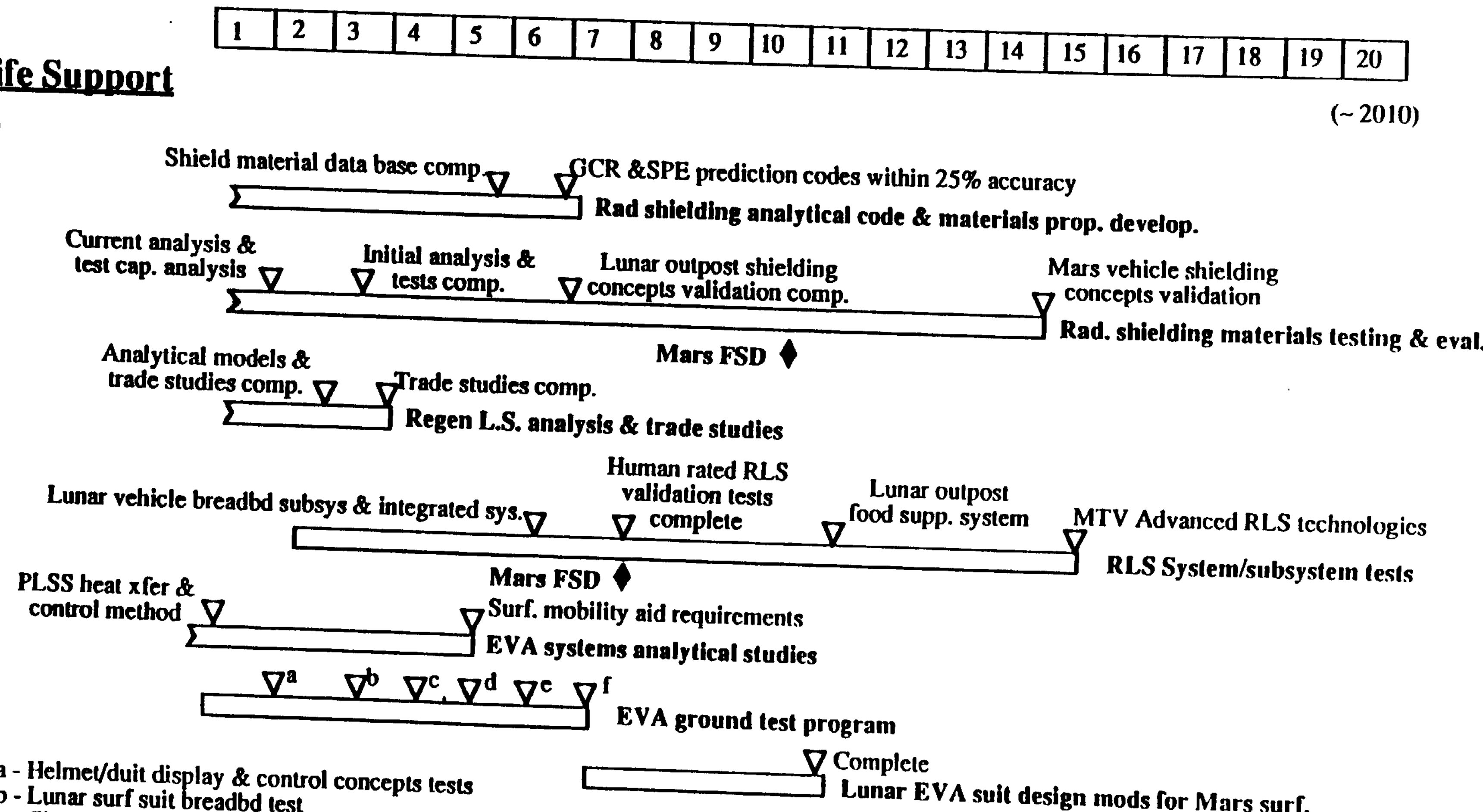
Life Support



a - Helmet/duit display & control concepts tests
b - Lunar surf suit breadbd test
c - Gloves & displays in simul. SSF environment
d - Regen. PLSS breadbd for lunar surf.
e - Verif tests of adv. dexterious gloves & disp.
f - Complete breadbd lunar EVA suit / simulated surf. cond.

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ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

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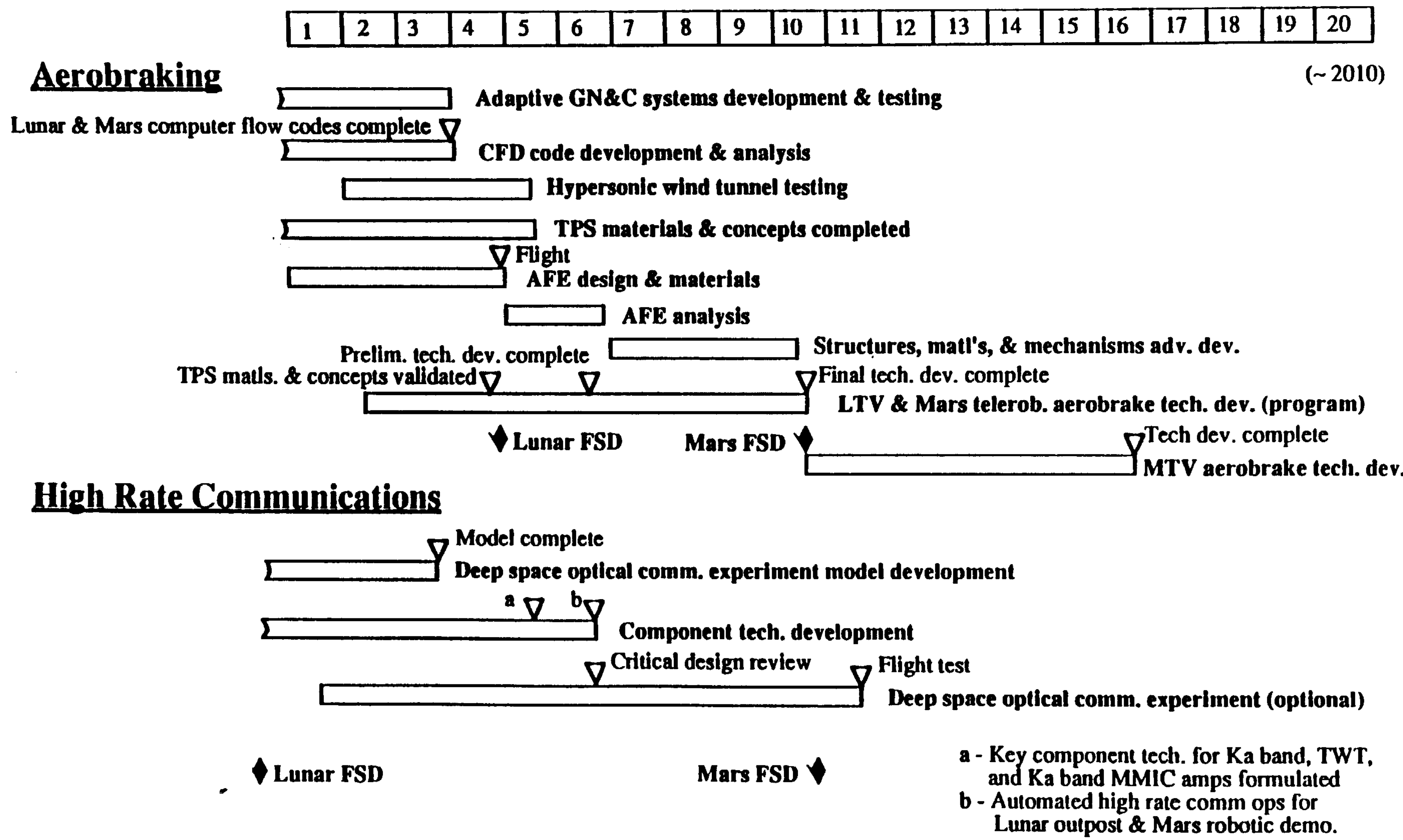

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ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

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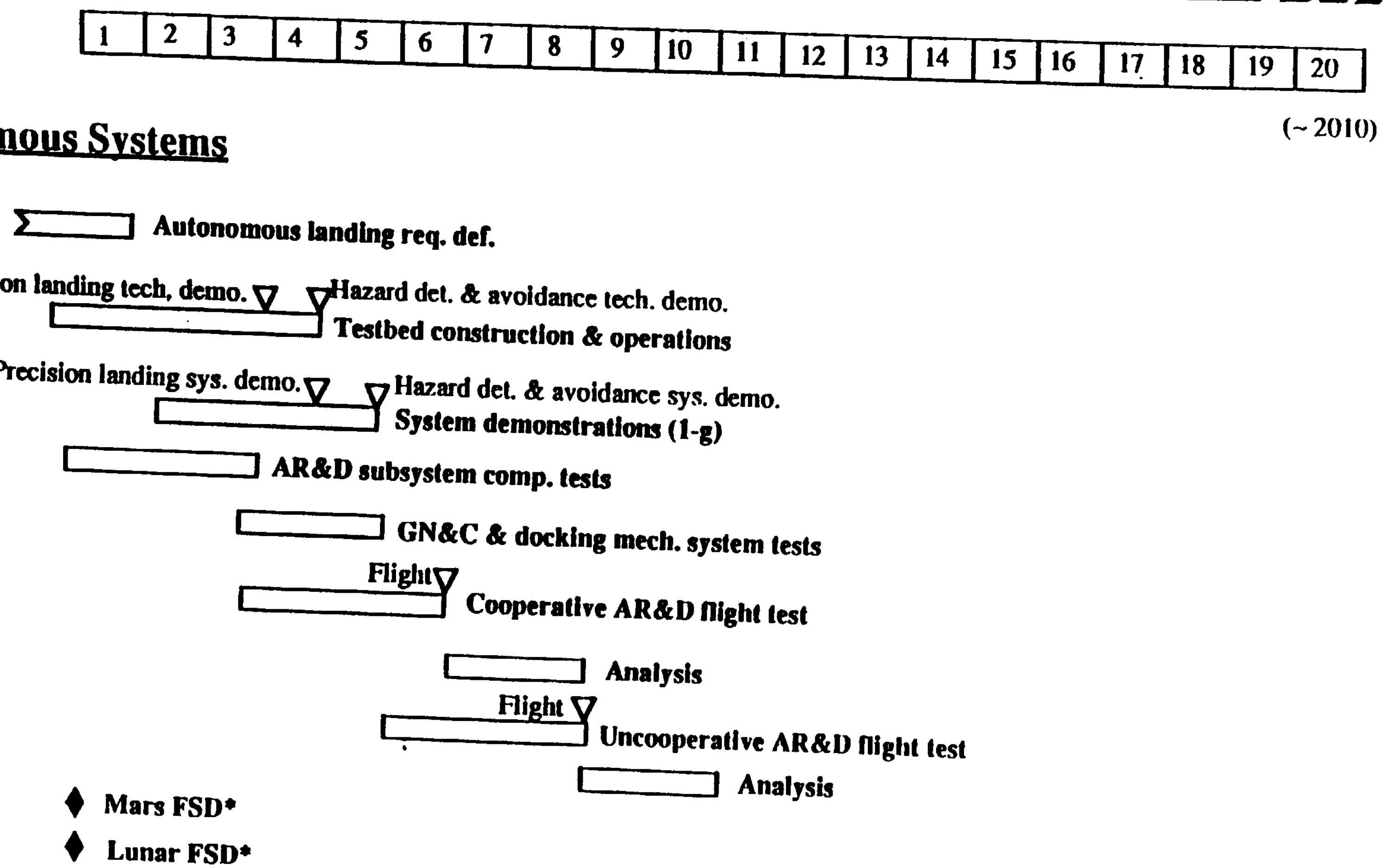
Autonomous Systems



* Technology should not present FSD threatening problems; current technologies adequate for minimum mission.

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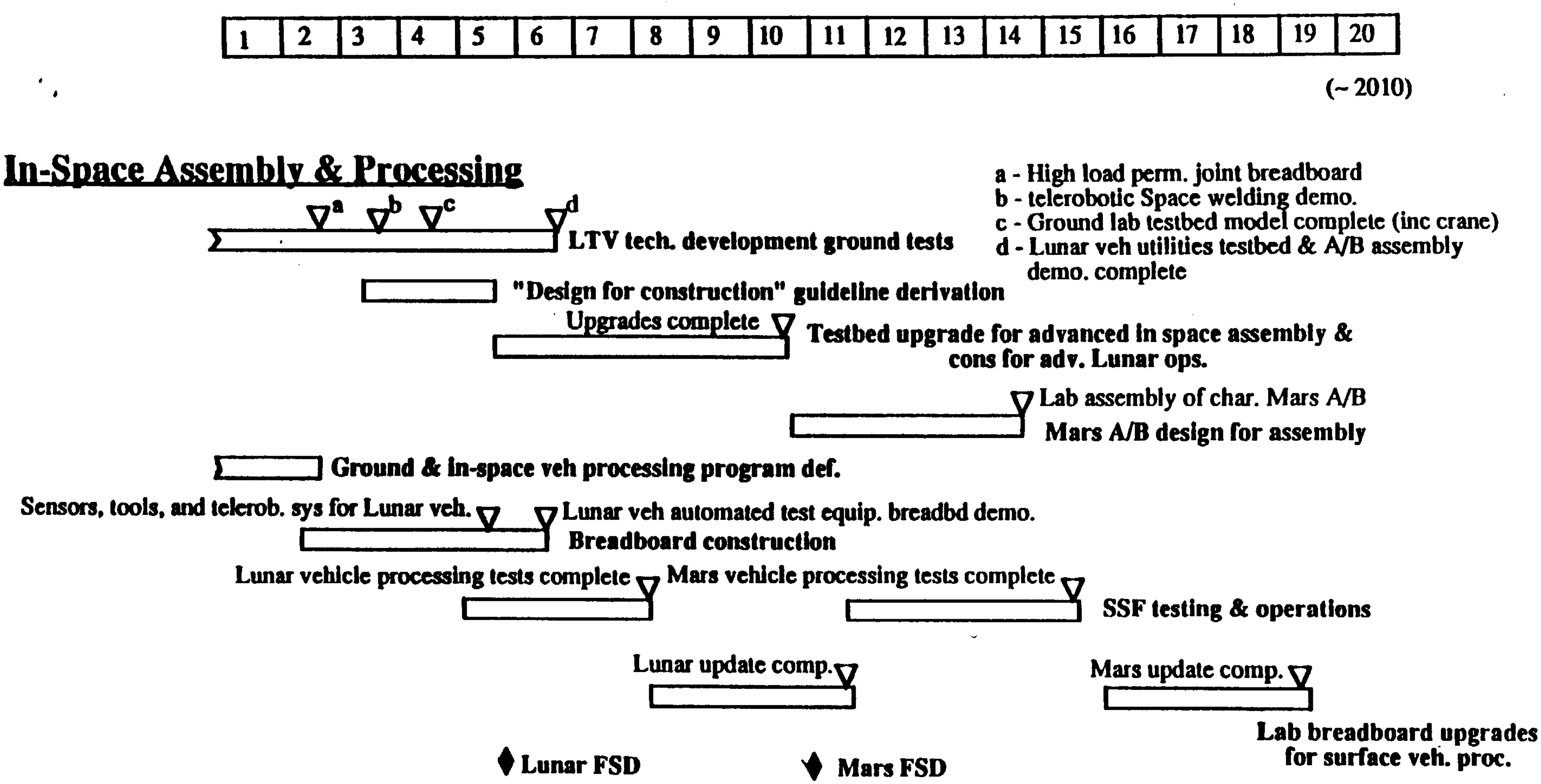
Preliminary SEI Technology Development Schedules (Cont.)

In-Space Assembly & Processing



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ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

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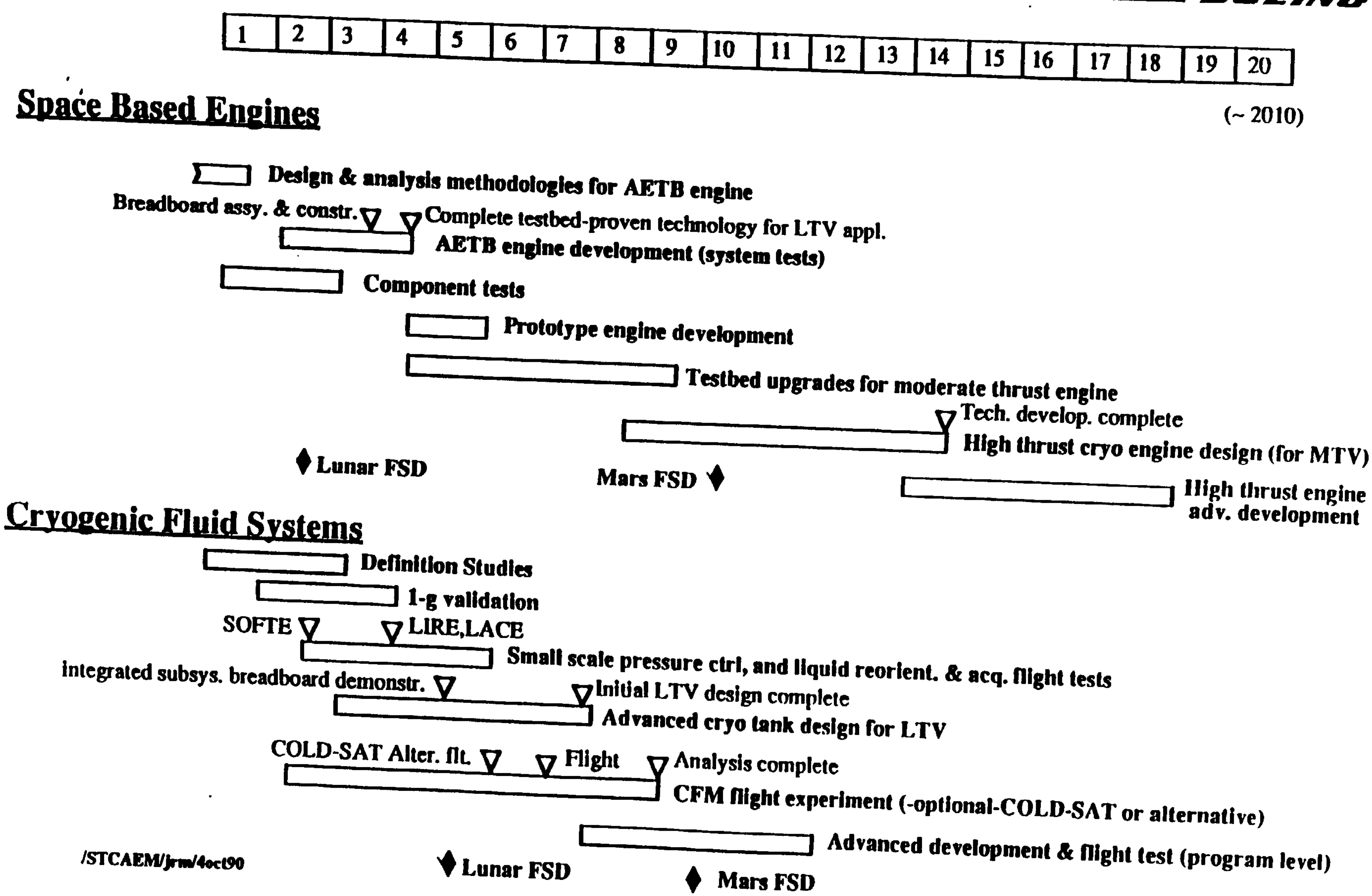

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Conclusions

Conclusions of the specific Level II trades were presented in the summary and with each trade description. The Level II trades, taken as a whole, illuminated sensitivities and leverages of systems and technologies. They provide the knowledge base as to how systems and technologies perform and how they can be integrated into mission systems. These trades, together with the vehicle concepts described in the other volumes of this document, are the basis for in-space transportation architecture synthesis.

A significant result of the study is that selection of a preferred mission architecture depends on overall mission objectives and activity level. More ambitious programs justify greater investment in technology advancement and in development of advanced systems. Because mission objectives and activity level have not yet been decided, and because of uncerteinties in costs and performance of many of the technologies, final architecture selections cannot be made now. A program strategy for technology advancement and initial development is needed.

Top-level recommendations for SEI program strategy were developed. These recognize that lunar and Mars exploration will start with modest objectives and evolve as a result of early mission achievements and scientific findings. An ambitious lunar program could significantly defer human Mars missions under likely funding constraints. Similarly, an ambitious Mars program could limit lunar activities to high-priority scientific objectives.

Our recommended SEI program strategy is evolutionary, allows changes in emphasis, and keeps options open. This strategy is expressed as architecture-level technology and program recommendations:

(1) Begin the manned lunar program with a tandem-direct expendable system.

(2) Invest in cryogenic storage and management technology and in a 30K-class advanced expander cryogenic engine with 10:1 or better throttling capability. These activities are candidates for advanced development.

(3) Baseline nuclear thermal rocket propulsion for Mars. Initiate a technology advancement program with emphasis on (a) high-performance fuels and (b) full-containment ground test facilities.

(4) Accelerate aerobraking technology for Mars aerocapture as a backup to the nuclear rocket, targeting a decision between the two in the 1996-2000 time frame.

(5) Perform aerobrake tests on the LTV booster, to put the technology on the shelf for Mars application.

(6) Designate solar-electric propulsion (SEP) as a "dark horse" for Mars transportation, and conduct a technology advancement effort aimed at removing the barriers to a high-performance, economic SEP system.

(7) Continue the present emphasis of the nuclear space power program on near-term systems applicable to planet surface power, but augment with (a) further studies to better understand the probable cost of nuclear power systems suitable for electric propulsion, and (b) modest funding of high-leverage high-performance power conversion technology.

Additional information on these recommendations may be found in the final technical report for the study.

Lunar Oxygen Delivery Mission Description

	Delta V m/sec	Mass Ratio	Propellant Used	Mass Remaining	
				71,729 kg.	(IMLEO) Resupply = 61,742
TLI	3110	1.95	34,938	36,791	LTV stage 7,902; Aerobrake 2,085; Hydrogen for LEV 23,085; Return propellant 1,203
Lunar pass & L2 arrive	330	1.073	2,516	34,275	
				76,851	(LTV start descent)
Descent from L2	2950	1.884	36,056	40,796	(Landed)
				274,298	(Liftoff)
Ascent to L2	2850	1.844	125,530	148,767	LEV Stage 16,639; Landing legs 1,224; Descent LO2 30,904; LO2 net to L2 95,000; LO2 tank 5,000
L2 depart & lunar pass	330	1.073	765	10,426	
Aerobrake to LEO	200	1.044	438	9,987	LTV stage 7,902; Aerobrake 2,085

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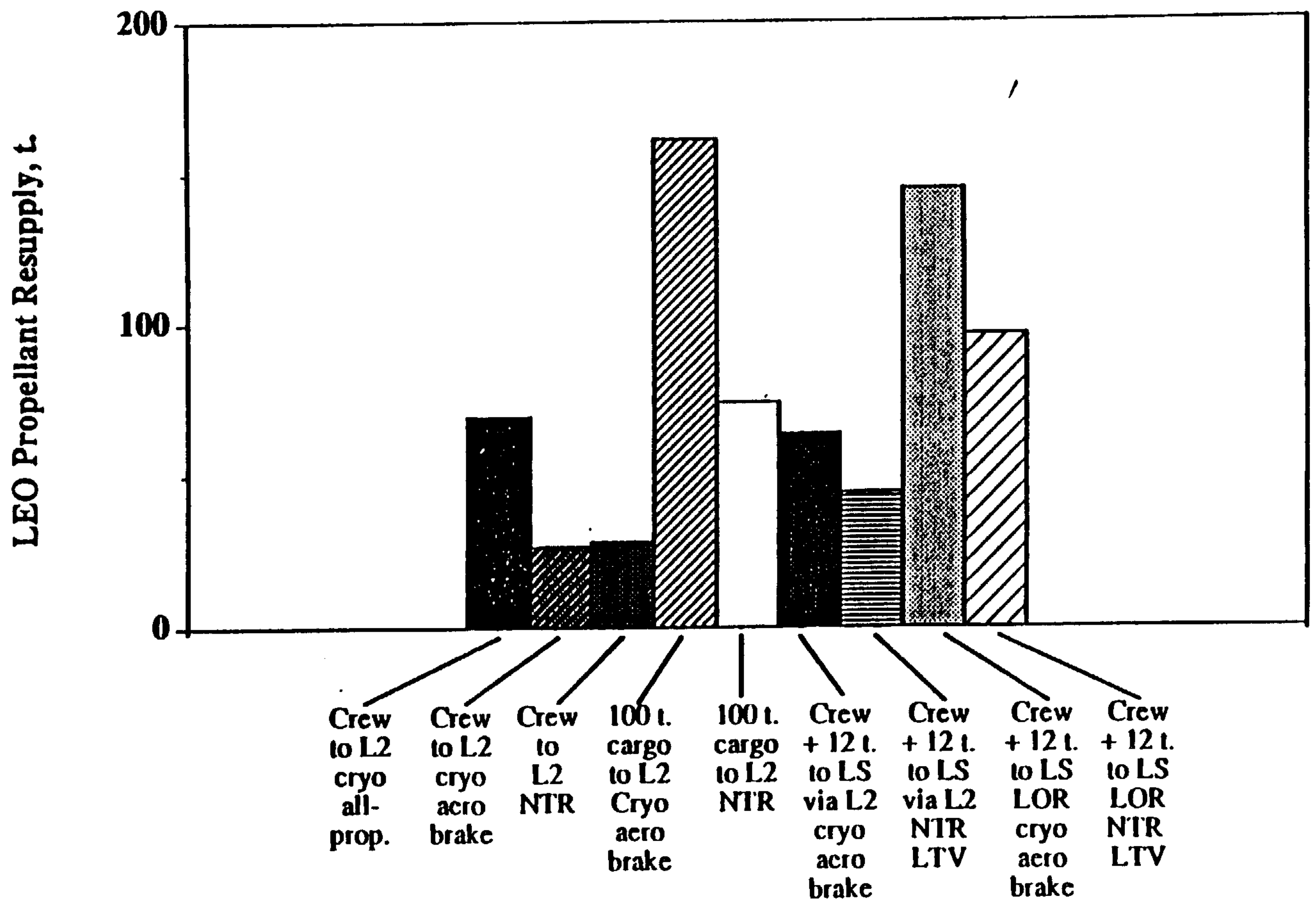
Cryo vs. NTR LTV Comparisons Preliminary Estimates



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Mars Summary

- More than 20 beneficial modes identified.

- Early Mars: Cryo all-propulsive (CAP), ECCV*, conjunction;
  NTR all-propulsive, conjunction or opposition;
  Cryo aerobraking opposition, ECCV;
  (possibly) Direct with Mars oxygen.

- High performance, late Mars or evolution:
  SEP or NEP;
  ISRU, moon or Mars or both;
  Combintations.

- Efficiency range 10:1 measured as RMLEO (resupply mass LEO).

- Reusable MEV/Mars propellant has significant leverage for high-performance options.

* Earth Crew Capture Vehicle, an Apollo-like capsule used for Earth entry and landing or aerocapture to LEO. The rest of the vehicle is expended.

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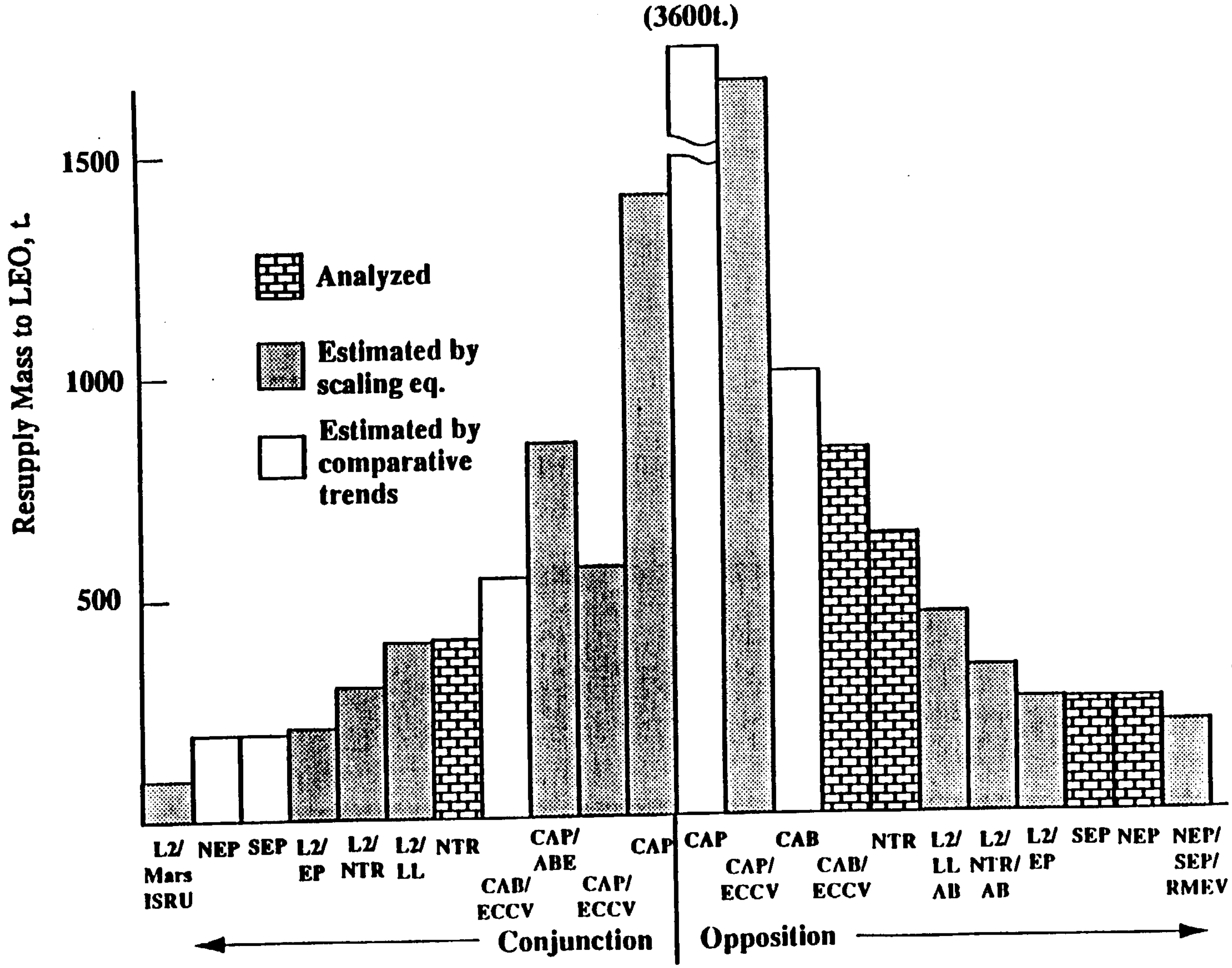
Comparison of Propulsion Options



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Conjunction vs. Opposition Mars Profiles

Opposition Advantages

- Shorter overall trip time, by at least a year.
- Transfer vehicle usually returns in time to be reused on next opportunity.
- Enables crew rotation/resupply mode with synodic period stay time.

Conjunction Advantages

- Lower energy; significantly less RMLEO unless very high Isp available.
- Venus swingby complexity not necessary.
- Long stay times at Mars.
- Shorter transfer times.
- Elliptic parking orbits can be optimized.

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Program Implementation Architectures

Architecture	Features	Aerobraking Functio Mars cap	Mars land	Earth cap/ lunar	Earth cap/ Mars	Earth entry*
Cryogenic/aerobraking	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	X	X	X	X	X
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.		X	X		X
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.		X	X		X
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.		X	X		X
L2 Based cryogenic/ aerobraking	L2-based operations; optional use of lunar oxygen.	**	X	X	X	X
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	X	X	X	X	
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report	***	X	X	X	X

Notes: * optional/emergency mode **opposition class only *** MEV-class crew taxi (not a large MTV)

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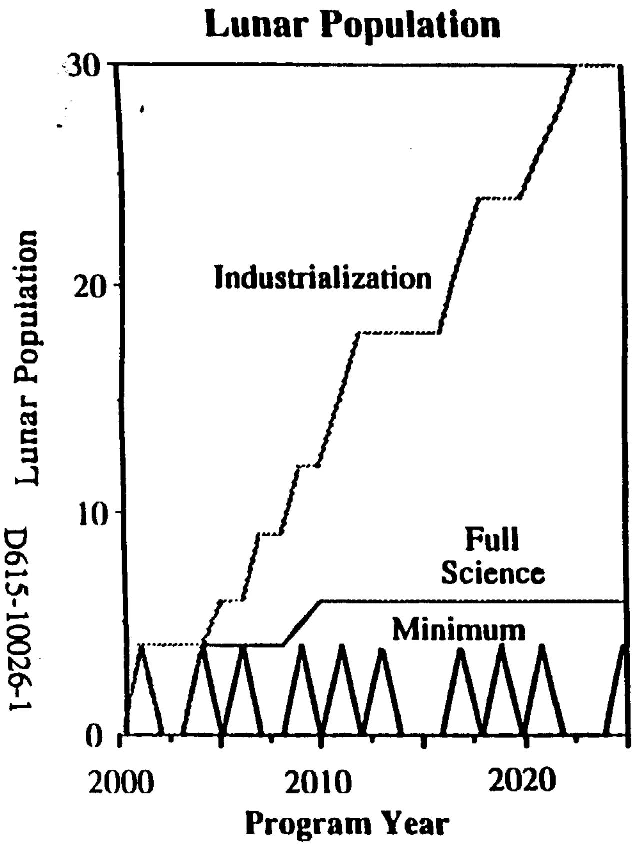
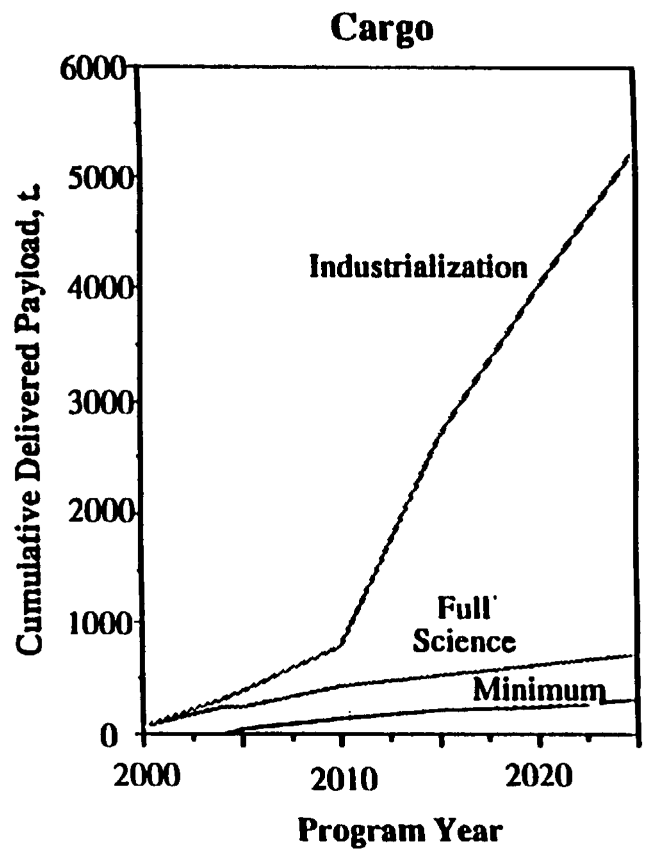
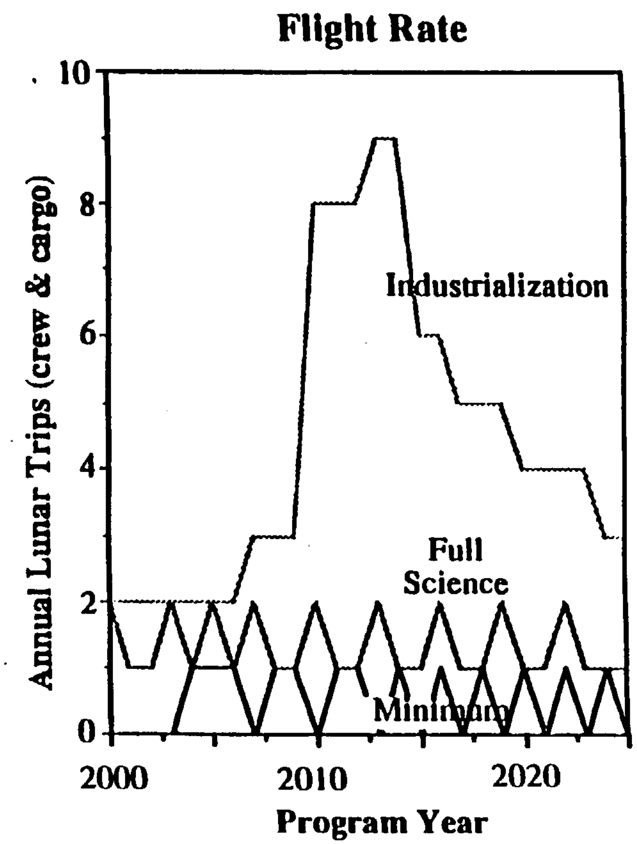
Lunar Implementations

	Minimum (a)	Full Science Menu (b)	Industrialization (c)
Purpose	Science station	Permanent science base	Industrialization
Surface Stay	45 days - overnight campsites	Permanent Yearly crew rotation	Permanent - stay time grows to 4 years.
Personnel	4 per visit	6 including scientists	More than 20
Beginning	2004	2000	2000
Surface Access	Local	Regional	Eventually global
CELSS	No	Yes	Yes
Surface Power	Solar/RFC	Nuclear	Nuclear + solar thermal process heat
Manufacturing	No	Experimental + lunar oxygen	Equivalent to ^3He production > 1 GWe + ~in-situ facilities
Propulsion	Cryo/aerobrake	Cryo/aerobrake + lunar oxygen	Cryo/aerobrake + lunar oxygen

Lunar Program Comparison







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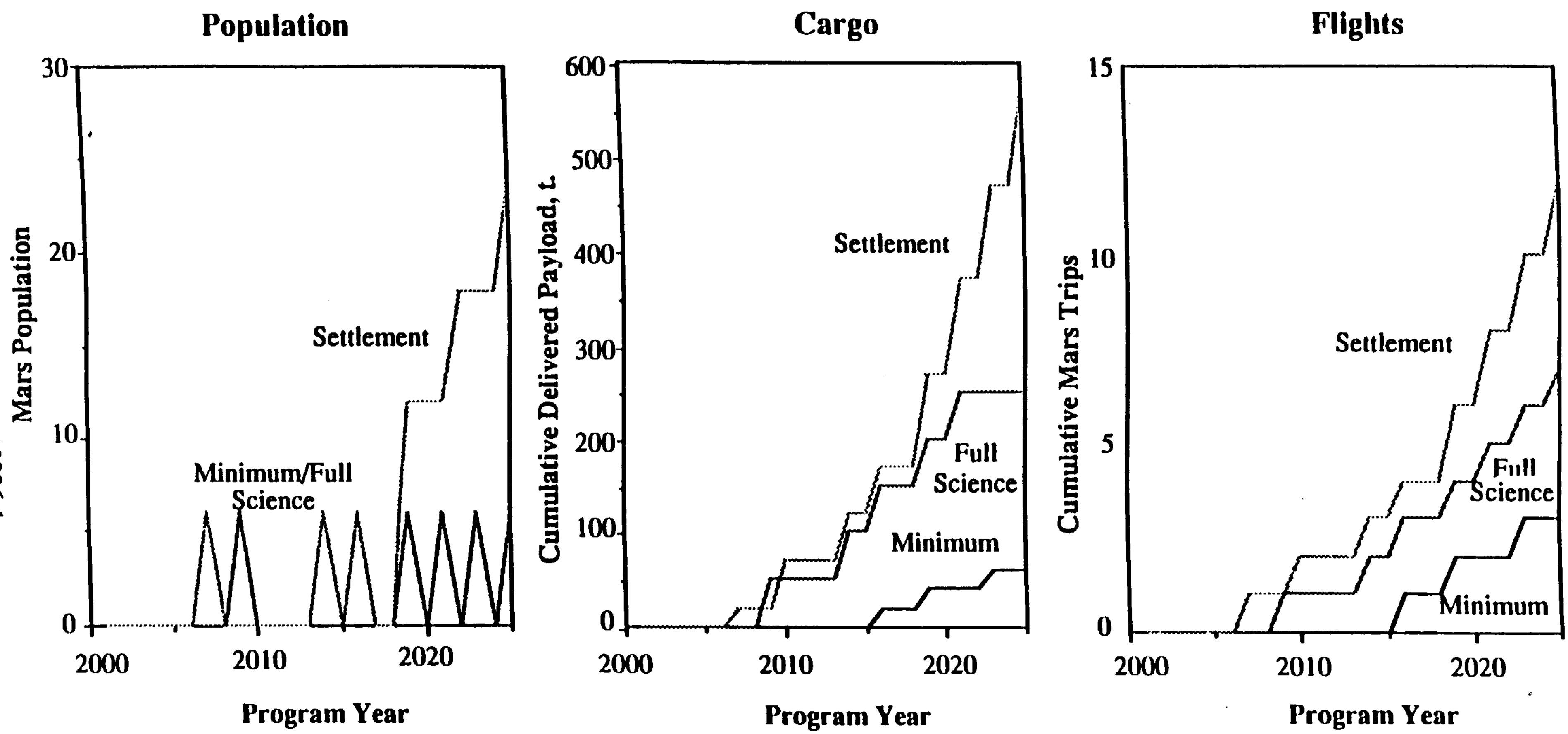
Mars Implementations

	Minimum (a)	Full Science Menu (b)	Settlement (c)
Purpose	Exploratory science	In-depth science	Science base leading to settlement
Surface Stay	30 days	1 year	3 years to permanent
Personnel	6 per visit	6 to 12	More than 20
Beginning	First landing 2016	First landing 2010	First landing (CAB) 2008
Surface Access	Local	Regional	Eventually global
CELSS	No	Partial	Yes
Surface Power	Solar/RFC	Solar/RFC	Nuclear
Manufacturing	No	No	Maximum practical self-sufficiency
Propulsion	Cryo/all-prop	Nuclear thermal rocket	Nuclear electric

Mars Program Comparisons



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Mars program scenarios

This document describes the strategy, scenario, and some assumptions for each of three Mars program architectures. These scenarios were used to develop mission manifests which Madison Research will as inputs to their program cost models. Several questions remain. Can the specified number of MEV flights in each case really provide all the equipment needed to support the specified crew for the stated duration? What is the cargo of the MEV's whose cargoes are currently listed as "stuff"? Despite these questions, enough detail is present to see the broad outlines of the programs, and thus to estimate their relative cost.

A general assumption has been that the basic MEV can carry six crew to the surface, but that it cannot support more than four for protracted periods, e.g. 30 days.

Minimum science program

The strategy of this program is to visit diverse Martian surface sites for brief human exploration, augmented by telerobotic reconnaissance. The program has three missions, identical but for the sites visited by each. A mission departs at every other conjunction opportunity, with the first departure in 2015. The propulsion technology is cryogenic all propulsive (CAP), and 100% of the hardware is expended in each mission.

The crew size is six per mission. All crew members visit the Martian surface, some perhaps twice: three or four crew shortly after arrival in Mars orbit, and three or four shortly before departure for Earth. The first surface crew stays 30 days, during which the crew explores locally using an unpressurized rover(s). After their return to orbit, the rover(s) telerobotically explores over a greater range, gathering samples. The second surface crew arrives in the most interesting area discovered by the rover(s). They conduct detailed local examinations and select the most valuable samples gathered by the rover(s) for return to Earth.

(Note that the above plan is not the absolute minimum. The cost of the mission could be reduced by using only one MEV. For maximum science return, the landing should still follow unmanned - preferably telerobotic - reconnaissance to select the best site. This can be accomplished by sending a teleoperated rover in a small landing vehicle, followed a year later by an MEV. The choice is whether to

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send all six crew, which may shorten the surface stay or call for a bigger (and more expensive) MEV, or to send only part of the crew, which could be bad for morale. The two-MEV scheme was chosen to avoid this dilemma and to provide redundancy.)

Full Science Scenario

The full science program strategy is to establish long-term bases for far-ranging surface exploration. The program has six missions before 2025. All follow conjunction trajectories, all use NTR propulsion (Isp 1050), and the MTV is reused.

Mission 1

The first mission departs in 2009 with a crew of six. Its surface itinerary is like a mission from the minimum science program: an early visit leaves a telerobotic rover for broad reconnaissance, and a second visit lands at the most interesting site. The site for the second visit is selected for its suitability as a base location. The second visit surveys the site, plants beacons, etc., and selects samples from the rover for return to Earth.

Mission 2

The second mission departs in 2013 with six crew, reusing the MTV from the first mission. It has three MEVs: one to deliver a hab module for six people, one to deliver power systems and consumables for the hab, and one to deliver the six crew and a pressurized scientific rover. The crew stays on the surface for about a year, deploying and outfitting the habitat, then exploring in the rover.

Mission 3

The third mission departs in 2015 with six crew, using an all-new MTV (the first MTV is still at Mars). The three MEV's carry an additional hab/lab module, consumables, a crane to unload modules from the landers, scientific equipment, and the crew. The crew stays down one year installing the new module, moving the old module from its MEV to the surface, mating the two modules, and exploring.

Mission 4

The fourth mission departs in 2019. It uses the original MTV, but has a new habitat for its twelve crew. One of its three MEV's carries a CELSS module to make the base more self-sufficient. Another MEV carries 6 crew and a new pressurized rover. The third MEV carries six more crew, consumables, and ISRU equipment. The crew install and operate the new module and equipment, as well as continue exploration and science.

Mission 5

Mission five departs in 2021. It uses the second MTV with a new hab for twelve crew. This mission has two regular MEV's and two mini-MEV's. The minis make two visits to the region of a second possible base. The first mini explores locally and leaves a telerobotic rover; the second visits the best site found and surveys it for a new base. The mini-MEV visits and the MEV surface stays can be timed to give each crew member a visit to both regions of the planet.

(This mission plan is questionable. Only mission 5 uses a mini-MEV; perhaps the mass savings do not justify development of this new piece of hardware for so limited a role.)

Mission 6

The sixth mission departs in 2023. It uses the first MTV with a new reactor (each reactor is good for three missions). There are twelve crew and four full size MEV's. Six crew and one MEV visit the original base, performing maintenance and continuing the regional exploration. The other six crew and the three other MEV's land at the new base site, essentially repeating mission 2 to establish a base at the new location and begin exploration.

Settlement Scenario

The settlement program strategy is to quickly establish the infrastructure needed to economically support large numbers of people on Mars. Science is supported as a secondary objective. The program includes seven missions before 2020, all using conjunction trajectories. The first two missions use expendable CAP technology, but all subsequent missions use NEP (Isp of 10,000 was assumed). Reusable MEVs (RMEV) are used beginning with the fifth mission.

Mission 1

The first mission departs in 2007. It carries a crew of six and two MEV's. The first MEV lands for 30 days with 3 or 4 crew and an unpressurized rover. The rover telerobotically explores the area after the MEV ascends. The second MEV lands later in the most promising base site uncovered by the rover. The second surface crew surveys the site, sets beacons, and collects samples from the rover.

Mission 2

The second mission departs in 2009 with six crew and three MEV's. One MEV carries a 6-person habitat. One carries a nuclear power plant, some consumables, and ISRU experiments. The third carries the crew and a pressurized bulldozer/backhoe that can serve as a 50 km rover. The crew installs the power plant (essential for frequent use of the bulldozer) and habitat, then begin civil engineering experiments and site preparation for later missions. They also carry out some exploration and science, time permitting. The crew stays on the surface for a year.

Mission 3

Mission three departs in 2011. The NEP vehicle carries twelve crew and four MEV's. Cargo delivered includes a constructible habitat, construction equipment, consumables, and a 1000 km rover. The rover can serve as a habitat for part of the crew until the constructible habitat is ready for occupants. The crew stays on the surface for over a year.

Mission 4

The fourth mission departs in 2013, delivering twelve crew and four MEV's aboard a second NEP vehicle (the first is still at Mars when mission four departs). This mission delivers a CELSS system, consumables and spares, and lots of ISRU equipment. The ISRU emphasis is on atmosphere distillation for nitrogen and water, on atmosphere cracking for oxygen and fuels, and on structural materials. The crew stays for over a year.

Mission 5

Mission five departs in 2015 with eighteen crew, three MEV's, and one RMEV. It delivers additional CELSS equipment, ISRU equipment, and an RMEV servicer. The crew installs the new equipment, refuels and reflies the RMEV, and leaves twelve people to stay on Mars until the next mission. The RMEV will have several flights left in its service life; it can be used as a rescue vehicle for long-range rover missions.

Mission 6

The first truly large cargo delivery, mission six delivers eighteen crew and two RMEV's. The RMEV's get five flights each, so the mission delivers 250 tons of cargo (currently not well specified) to the surface. The twelve crew left from the previous mission return to Earth, and a new set of twelve stay over until the next mission.

Mission 7

Mission seven is the first "steady state" mission, requiring no new equipment for the MTV besides the usual replacement of thrusters, propellant, and consumables. The mission requires only seven HLLV launches of hardware. It delivers 250 tons and eighteen crew to Mars, expends two RMEV's, and returns twelve people to Earth (only six, perhaps less, are needed to safely operate the MTV). Replacing a reactor on every third flight, this pattern can be maintained indefinitely, settling Mars at a rate of six to twelve more people every two years.

(The tremendous cargo capacity of the last two missions suggests that the scenario should be replanned for earlier use of RMEV's and *in situ* propellant. This could greatly accelerate the arrival of large numbers of crew and equipment.)

Lunar/Mars Manifest Worksheet

Minimum Scenario

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (Production Rqts)	Propellant to LEO	Total to LEO	Tankers	ETO Launches	Moon/Mars Pop.
2015, 2019, 2023	explore	CAP conj.	85.8	TMI Strap-on x 5		394.67	428.9		5	6
			97.1	TMI core		78.9	97.1		1	
			73.1	MEV x 2	2 unpress. rovers, science eqpt.		146.2		2	
			60.2	Hab for 6, ECCV			60.2		1	
			65.0	MOC core		55.6	65.0		1	
			61.1	MOC strap-on		55.6	61.1		1	
			100.0	top-off prop.		90.0	100.0	1		

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Lunar/Mars Manifest Worksheet

Full Science Scenario

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (Production Rqts)	Propellant to LEO	Total to LEO	Tankers	ETO Launches	Moon/Mars Pop.
2009	scout base sites	NTR conj.	95.4	Truss, PBR reactor #1, aft tank, prop. for all but TMI		54.6	95.4		1	6
			92.8	TMI tanks x 2		159.0	185.6		2	
			73.1	MEV x 2	2 unpress. rovers, science eqpt.		146.2		2	
			53.1	Hab for 6 (#1)			53.1		1	
				top-off prop.		90		1		
2011		LTV		HEO>LEO prop		89		1		
2013	early base	NTR conj.	100	TMI tanks x 2, partly full		167.2	200		2	
			73.1+	MEV x 3, hab consumables	6-crew surface hab, power sys, press. rover		235.4		3	
				top-off prop.		90		1		
2015		LTV		HEO>LEO prop		89		1		

bjt/STCAEM/Oct 3, 1990/p1/3

Lunar/Mars Manifest Worksheet

Full Science Scenario

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (Production Rqts)	Propellant to LEO	Total to LEO	Tankers	ETO Launches	Moon/Mars Pop.
2015	expand base	NTR conj.	100	Truss, PBR reactor #2, aft tank, prop. for all but TMI		59.2	100		1	6
			100	TMI tanks x 2		167.2	200		2	
			73.1	MEV x 3	crane, 6-crew surface hab add-on		219.3		3	
			53.1	MTVHab for 6 (#2)			53.1		1	
				top-off prop.		90		1		
2017		LTV		HEO>LEO prop.		89		1		
2019 (reactor #1)	explore, consolidate	NTR conj.	100	TMI tanks x 2	-	163.4	200		2	12
			73.1	MEV x 3	CELSS, press. rover, ISRU experiments		219.3		3	
			99.9	12-crew MTV hab #3			99.9		1	
				top-off prop.		180		2		
2021		LTV		HEO>LEO prop.		89		1		
					Replace LTV				1	

bjt/STCAEM/Oct 3, 1990/p2/3

Lunar/Mars Manifest Worksheet

Full Science Scenario

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (Production Rqts)	Propellant to LEO	Total to LEO	Tankers	ETO Launches	Moon/Mars Pop.
2021 (reactor #2, hab #4)	explore, scout new base site	NTR conj.	100.0	TMI tanks x 2		161.4	200.0		2	12
			73.1	MEV x 2	?		146.2		2	
			96.0	mini-MEV x 2	unpress. rover x 2		96.0		1	
			99.9	12-crew MTV hab #4			99.9		1	
				top-off prop.		270.0		3		
2023		LTV		HEO>LEO prop.		89.0		1		
2023 (reactor #3, hab #3)	begin 2nd base, use old base	NTR conj.	100.0	TMI tanks x 2		157.2	200.0		2	12
			>73.1	MEV x 4, NTR reactor #3	6-crew hab, power sys., pressurized rover, MTV consumables		334.2		4	
				top-off prop.		270.0		3		
2025		LTV		HEO>LEO prop.		89.0		1		

bjt/STCAEM/Oct 3, 1990/p3/3

Lunar/Mars Manifest Worksheet

Settlement Scenario

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (Production Rqts)	Propellant to LEO	Total to LEO	Tankers	ETO Launches	Moon/Mars Pop.
2007	scout base site	CAP conj.	96.2	TMI Strap-on x 5		441.5	481.2		5	6
			100.0	TMI core (partly fueled)		80.8	100.0		1	
			73.1	MEV x 2	2 unpress. rovers, science eqpt., beacons		146.2		2	
			60.2	Hab for 6, ECCV			60.2		1	
			65.0	MOC core		61.0	71.0		1	
			61.1	MOC strap-on		61.0	67.1		1	
				top-off prop.		90.0		1		
2009	early base	CAP conj.	95.2	TMI Strap-on x 7		608.8	666.7		7	6
			100.0	TMI core (partly fueled)		80.5	100.0		1	
			73.1	MEV x 3	6-crew hab, reactor, bulldozer/rover, ISRU experiments		219.3		3	
			60.2	Hab for 6, ECCV			60.2		1	
			88.6	MOC core		77.2	88.6		1	
			84.7	MOC strap-on		77.2	84.7		1	
				top-off prop.		90.0		1		

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Lunar/Mars Manifest Worksheet

Settlement Scenario

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (Production Rqts)	Propellant to LEO	Total to LEO	Tankers	ETO Launches	Moon/Mars Pop.
2011	expand base	NEP conj.	98.9	reactor #1, generators			98.9		1	12
			>73.1	MEV x 4, NEP power dist & control, prpllnt	constructible hab, construction equipmt., 1000 km rover, consum	21.8	350.0		4	
			~42.9	rest of NEP vehicle	radiators, att. cntl., truss, thrusters, etc.		85.8		2	
			100.0	prop. tanks x 6		181.8	200.0		2	
			99.9	Hab for 12 (#1)			99.9		1	
2013	establish self-sufficiency	NEP conj.	98.9	reactor #2, generators			98.9		1	12
			>73.1	MEV x 4, NEP power dist & control, prpllnt	CELSS, ISRU, science eqpt., consumables	21.8	350.0		4	
			~42.9	rest of NEP vehicle	radiators, att. cntl., truss, thrusters, etc.		85.8		2	
			100.0	prop. tanks x 6		181.8	200.0		2	
			99.9	Hab for 12 (#2)			99.9			

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Lunar/Mars Manifest Worksheet

Settlement Scenario

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (Production Rqts)	Propellant to LEO	Total to LEO	Tankers	ETO Launches	Moon/Mars Pop.
2015	first 3-year stay (18 crew, 12 stay over), test RMEV	NEP conj. (rctr 1, hab 1)	94.0	RMEV, 1/4 of thrusters	CELSS		94.0		1	18
			83.8	MEV x 3, consumables for MTV hab	consumables, RMEV servicer, ISRU		251.4		3	
			100.0	prop. tanks x 9, 1/4 of thrusters		248.3	286.8		3	
			86.7	Hab for 6, 1/2 of thrusters			86.7		1	
2017	retrieve 12 from last trip, leave another 12	NEP conj. (rctr 2, hab 2)	94.0	RMEV x 2, 1/2 of thrusters	constr. eqpt., new reactor, outfitting		188.0		2	18
			100.0		RMEV pallets: RMEV servicer, consum, CELSS, const. equip.		200.0		2	
			100.0	prop. tanks x 8		242.6	266.9		3	
			86.7	Hab for 6, 1/2 of thrusters			86.7		1	
2019+	retrieve 12 from last trip, leave another 18	NEP conj. (rctr 1, hab 1)	94.0	RMEV x 2, 1/2 of thrusters	constr. eqpt., outfitting		188.0		2	18
			100.0		RMEV pallets: const. hab, outfitting, science rover		200.0		2	
			100.0	prop. tanks x 8, 1/2 of thrusters		242.6	294.3		3	

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Program Scales For Transportation Architecture Analysis

Small Scale

	Lunar	Mars
Purpose	Near-side science station	Exploration Sample collection
Surface Stay	45 days Overnight campsites (LTV with shielding)	20 days Live in the landers
Beginning	2004	2015
Frequency	Flights every other year	Flights every other year (opposition or conjunction)
Growth	Pressurized rovers Flyers	Unpressurized rovers
Precursors	Lunar observer Early manned visit	2 Mars Observers Long-lived rovers

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Program Scales For Transportation Architecture Analysis

Medium Scale

	Lunar	Mars
Purpose	Permanent science base	Permanent science base
Crew	6 member science	6 member science
	(additional personnel as required for base maintainence and ISRU processes)	
Rotation	Yearly	Each opposition opportunity (possible unoccupied periods)
Beginning	2000 (no SSF support)	2010
Access	Global	Regional Pressurized rovers
CELSS	Yes	Yes (possible problems due to unoccupied periods)
Precursors	1 manned visit	>3 manned visits
Other	Nuclear surface power In-situ science Science facilities >10 km from habitat site	Robotic cargo flights

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Program Scales For Transportation Architecture Analysis

Large Scale
(builds on Medium Scale scenario)

	Lunar	Mars
Purpose	Settlement and industrialization	Settlement
Rotation	2 Years	Change-out opportunity 3 to 5 years
Beginning	2000 (no SSF support)	First Mission TBD (12 people each flight starting in 2020)
Access	Global	Global by 2025 (if possible)
Manufacturing	Equivalent to He production (>1 GWe) >80% in-situ facility production	80% in-situ facility production

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Representative Lunar Base Applications

	Small	Medium	Large
Purpose	Science Station	Permanent Base	Settlement Industrialization
Surface Stay	45 days Overnight Campsites	Permanent Yearly crew rotation	Permanent 2-year rotation
Personnel	4 per visit	6 member science (additional personnel as req'd for base maintainence and ISRU processes)	TBD
Beginning	2004	2000 (no SSF support)	2000
Access	Regional	Global	Global
CELSS	No	Yes	Yes
Surface Power	Solar / RFC	Nuclear	Nuclear
Manufacturing	No	Experimental only except for lunar/Mars oxygen production	Equivalent to He production (>1 GWe) >80% in-situ facility production Solar powered facilities
Precursors	Lunar Observer Early manned visit	1 manned visit	Medium scenario

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Lunar Mission - Small Scale

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Small Lunar Automated Telescope	400	1.5	TBD	Automated
* Lunar Gravity Wave Detector	5000	20	TBD	Set up components
* Crater Telescope	500	2	0.5/Element	2/EVA;2 crew/telescope; set up: 2hrs/telescope; 2 crew; Maintenance: 1 EVA/yr
* UV-Visible Interferometer (4 Elements)	4000	30	0.5/Element	
* Sub-millimeter (IR) Array (1 Element)	6000	50	0.5/Element	
* Lunar Optical Interferometer				Construction of Moon facilities; continuous ground operation
Initial 3 Telescopes	4300	130	2.5	
Final 12 Telescopes (by mission end)	16000	470	8.5	
TOTAL	15900	103.5	TBD	

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Lunar Mission - Small Scale

LIFE SCIENCES

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Exobiology (Biostack, Asceptic Samplers)	50	TBD	TBD	Included in geology EVA
* Biological Sample Management Facility	125	0.5	0.25	TBD
* Bioregenerative Life Support Facility	300	0.5	0.3	TBD
* Biolab - ESA	272	0.9	1.13	TBD
* Life Science Equip. NASDA	590	1.75	1.95	TBD
* CSA Life Science	42	0.31	0.551	TBD

Lunar Mission - Small Scale

PLANETARY SCIENCE

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Lunar Astronaut Field Package	300	1	On-board	Direct Operation of some equipment
* Lunar Environment Station	100	TBD	On-board	Some station deployment
* Lunar Sample Return Containers (10 Sets)	200			
(Delivery)		0.3	0	Fill and stow for return
(Return)		1.2	0	
* Lunar Penetrator with Descent Imaging	100	1	On-board	Command generation Data relay
* Laser Ranging Retro-reflector	50	TBD	0.1	Emplaced during geology EVA
* Particles and Field Stations (2 stations and 1 Particle detector)	1000	TBD	RTG	Maintenance: 4 hrs/yr
TOTAL	4200	TBD	TBD	

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Lunar Mission - Small Scale

TRANSPORTATION REQUIREMENTS

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Campsite	TBD	TBD	TBD	TBD
* LEV	TBD	TBD		
* Rovers				
Presssurized	TBD	TBD		
Unpressurized	TBD	TBD		
Robotic	800	16		
* Flyers				
Pressurized (25 t. payload)	55 t.	Mini MEV		
Unpressurized (t. payload)	10 t.	15' x 30'		

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ADVANCED CIVIL SPACE SYSTEMS

Lunar Mission - Medium and Large Scales

BOEING

ASTRONOMY

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Small Lunar Automated Telescope	400	1.5	TBD	Automated
* Lunar Gravity Wave Detector	5000	20	TBD	Set up components
* Crater Telescope	500	2	0.5/Element	2/EVA;2 crew/telescope; set up: 2hrs/telescope; 2 crew; Maintenance: 1 EVA/yr
* UV-Visible Interferometer (4 Elements)	4000	30	0.5/Element	
* Sub-millimeter (IR) Array (1 Element)	6000	50	0.5/Element	

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Lunar Mission - Medium and Large Scales

ASTRONOMY

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Large Optical Telescope				
4 - m Configuration	15000	300	1.5	All of these listed here require facility construction, continuous ground control, and maintenance.
16 - m Configuration	42000	TBD	5	
* Lunar Transit Telescope				
0.8 - m Aperture	1000	TBD	0.35	
1.8 - m Aperture	1250	8	0.35	
3.5 - m Aperture	4800	TBD	0.35	
* Optical Interferometer				Each system is large, with different baseline lenghts.
Initial 3 Telescopes	4300	130	2.5	
Final 12 Telescopes	16000	470	8.5	
* Nearside VLF Imaging Array	-500	TBD	TBD	
* Farside VLF Interferometer	1000	TBD	1	
* Sub Millimeter Interferometer	12000	700	20	
TOTAL	111750	TBD	TBD	

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Lunar Mission - Medium and Large Scales

LIFE SCIENCES

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* 1.8-m Centrifuge Facility	1480	2	2.4	Included in geology EVA
* Habitat Holding System	500	1	1.9	TBD
* Biologincal Sample Management Facility	125	0.5	0.25	
* Bioinstrumentaltion & Physiologicl Monitoring Fac.	750	2.5	0.8	TBD
* Analytical Instruments Fac.	250	0.75	0.65	
* Life Science Lab Support Equipment Facility	325	1	0.6	TBD
* Centralized Life Science Computer Fac.	50	0.25	0.4	TBD
* Bioregenerative Life Support Façility	300	0.5	0.5	
* Exobiology Facility	250	0.5	0.5	

Lunar Mission - Medium and Large Scales

LIFE SCIENCES

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Commercial Life Sciences Facility	300	1	1	
* ESA equipment	580	1.9	3	
* NASDA Equipment	2713	8.825	10.7	
* CSA Equipment	172	1.64	2.451	
TOTAL	7695	47.115	32.351	

Lunar Mission - Medium and Large Scales

PLANETARY SCIENCE

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Lunar Astronaut Field Package	300	1	On-board	Direct Operation of some equipment
* Lunar Environment Station	100	TBD	On-board	Some station deployment
* Lunar Sample Return Containers (10 Sets)	200			Fill and stow for return
(Delivery)		0.3	0	
(Return)		1.2	0	
* Lunar Penetrator with Descent Imaging	100	1	On-board	Command generation Data relay
* Laser Ranging Retro-reflector	50	TBD	0.1	Emplaced during geology EVA
* Particles and Field Stations (2 stations and 1 Particle detector)	1000	TBD	RTG	Maintenance: 4 hrs/yr

Lunar Mission - Medium and Large Scales

PLANETARY SCIENCE

	Mass (kg)	Volume (m^3)	Power (kW)
* Geoscience Laboratory Intstruments			
Emplacement Phase	40	0.04	0.056
Consolidation Phase	24	0.05	0.096
* Lunar 10 - m Drill	200	0.1	5.02
* Lunar Deep Drill	20000	200	500
TOTAL	24464	TBD	505.272
* Analytical Science Lab. Equipment	1000	TBD	10
* Geologic Exploration Equipment	100	TBD	TBD
* Drilling Equipment	1000	TBD	10/3 days
* Geophysical Stations (6 @ 100 kg)	600	TBD	RTG
* Portable Geophysical Package	100	TBD	TBD

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Lunar Mission - Medium and Large Scales

PLANETARY APPLICATIONS

Medium Scale:

Lunar LOX

- Baseline: *Robotic Lunar Surface Operations* (begin on p. 73)
- Production of 100 tons/year Need to determine production rate as to scale plant
- Time AFL to habitability: 1.5 years

Large Scale:

1 GWe He Power Plant

- Base on *Report of NASA Lunar Energy Enterprise Case Study Task Force*
- Determine amount of He required
- Etc.

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Lunar Mission - Medium and Large Scales

TRANSPORTATION REQUIREMENTS

	Mass (kg)	Volume (m^3)	Power (kW)
* LTV			
* LEV			
*Habitation Modules	TBD	TBD	TBD
*Rovers	20 t + equip.		
Pressurized			
Unpressurized			
*Flyer			
Unpressurized (t payload)	10000	15'x30'	
Pressurized (25 t payload)	55000	mini MEV	

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Mars Mission - Small Scale

PLANETARY SCIENCE

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Mars Astronaut Field Package	300	1	Battery	Direct Operation
* Mars Environmental Station	100	1	TBD	Station deployment
* Mars Drill				Site selection
Astronaut-held	50	0.1	TBD	Operation
Automated lander/rover	30	0.1	0.5	N/A
* Mars Balloons	200	1	0.005	Inflation and launch
* Imaging Impactors (12 @ 50 kg and 0.2 m^3 each)	600	2.4	On-board	TBD
* Mars Multipurpose Tower	200	1	0.056	EVA for erection, guying, maintenance
*Venus Atmospheric Probe	300	5	TBD	Predeploy checkouts and deployment
TOTAL	1780	11.6		

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Mars Mission - Medium and Large Scales

ASTRONOMY

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Solar Flare Monitors				
Manned Interplanetary Veh.	300	TBD	TBD	Deployment
Dedicated unmannned launch	300+tbd	TBD	TBD	N/A
Micro-spacecraft	50	TBD	TBD	possible
* Fields and Particles	50	0.1	22	None

Mars Mission - Medium and Large Scales

PLANETARY SCIENCE

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Geologic Exploration Equipment	100	TBD	TBD	TBD
* Geophysical/Meteorology Station (10 @ 100 kg each)	1000	TBD	RTG	TBD
* Portable Geophysical Traverse Package	100	TBD	RTG	2 EVAs to emplace; used during traverses
* Analytical Science Lab. Equipment	5-900	TBD	2-10	Pressurized
* Preliminary Biomedical Lab.	5-900	TBD	0.5-1	IVA: 6-10 hrs/week
* Plant/animal /microbe Instruments	2300	TBD	2	IVA: 6 hrs/.week
* Mars Resource Experiments (includes Water Extraction)	600	TBD	0.5	IVA: 8 hrs/week
* Materials Research Instruments	1000	TBD	1	IVA: 4 hrs/week

Mars Mission - Medium and Large Scales

PLANETARY SCIENCE

	Mass (kg)	Volume (m^3)	Power (kW)	Human Interaction
* Mars Astronaut Field Package	300	1	Batteries	Direct Operation
* Mars Environmental Station	100	1	TBD	Station deployment
* Mars Drill				
Astronaut-held	50	0.1	TBD	Site selection Operation
Automated lander/rover	30	0.1	0.5	N/A
Astronaut emplaced	500	10	5.5	
* Mars Balloons	200	1	0.005	Inflation and launch
* Imaging Impactors (12 @ 50 kg and 0.2 m^3 each)	600	2.4	On-board	TBD
* Mars Multipurpose Tower	200	1	0.056	EVA for erection, guying, maintenance
*Venus Atmospheric Probe	300	5	TBD	Predeploy checkouts and deployment
* Geoscience Laboratory Equipment	120	0.32	0.253	Direct operation in pressurized lab.
* Mars Airplane (from orbit)	150	TBD	TBD	TBD
TOTAL	2050	TBD	TBD	

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Lunar Flights - Small Scale

ASSUMPTIONS:

- Flights approximately every other year.
- Use tandem LTVs - LEV hasn't been developed.
- Scientific outpost
- Overnight stays in a campsite
- Crew size of 4

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Lunar Flights - Small Scale

Flight Number	Year	Mission Description
1	2004	Crew of 4 - 4 to 6 days on Lunar Surface - Direct flight using tandem LTVs
2	2005	Campsite delivery.
3	2006	Crew of 4 - First overnight stay; - Tandem LTV fight to Lunar Surface; - Science based mission.
4	2008	Cargo delivery.
5	2009	Crew of 4 - Overnight stay; - Science based mission.
6	2011	Crew of 4 - Overnight stay
7	2013	Crew of 4 - Overnight stay
8	2015	Science cargo delivery.
9	2017	Crew of 4 - Science based mission.
10	2019	Crew of 4 - Another overnight stay.
11	2021	Crew of 4 - Overnight stay;
12	2023	Cargo delivery.
13	2025	Crew of 4 - Overnight, science mission.

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Lunar Manifest Worksheet - Small Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2004, 2011, 2019	Crew	LTVT Dir/fo	10 t	2 LTVs, 1 Brake, 1 ACRV, Crew Cab, Landing Legs	2 LTVs, 1 Brake, 1 ACRV, Crew Cab, LTV Landing Legs	209 t	261 t	3	4 HLLV 1 Crew	4 Crew, 4 - 6 days for 1st mission, 40 days on later missions
2005	Campsite Delivery	LTVT Dir,exp	25 t + Camp-site	2 LTVs, 1 Brake Landing Legs	1 LTV, Campsite, LTV Landing Legs, Spare Brake (reused LTV and Brake)	206 t	290 t	3	3 HLLV	N/A
2006, 2013, 2021	Crew	LTVT Dir/fo	10 t	2 LTVs, 1 Brake 1 ACRV, Crew Cab, Landing Legs	Spare LTV, 1 ACRV, Landing Legs, Crew Cab (reused LTVand Brake)	213 t	267 t	3	3 HLLV 1 Crew	4 Crew, 40 days on LS
2008, 2015, 2023	Science Cargo	LTVT Dir,exp	55 t	2 LTVs, 1 Brake Landing Legs	Science Cargo, 1 LTV, Landing Legs (reused LTV and Brake)	206 t	290 t	3	3 HLLV	N/A
2009, 2017, 2025	Crew	LTVT Dir/fo	10 t	2 LTVs, 1 Brake 1 ACRV, Crew Cab, Landing Legs	1 LTV, 1 ACRV, Landing Legs, Crew Cab (reused LTV and refurbished Brake)	213 t	267 t	3	3 HLLV 1 Crew	4 Crew, 40 days on LS

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Lunar Flights - Medium Scale

ASSUMPTIONS:

- One crew flight per year.
- Cargo flight rates will vary.
- Flights begin in 2000.
- Initially no Space Station support.
- By 2010, approximately 700 tonnes are needed on the lunar surface. This includes the LLOX plant and base construction materials.
- A campsite will be used to house the crew until the base is completed.
- Crew flights initially have 4 crew members until 2010. Crew size increases to 6.
- Lunar Oxygen will be used for ascent and descent of the LEV beginning in 2010.

Lunar Flights - Medium Scale

Flight Number	Year	Mission Description
1	2000	Crew Sortie
2	2000	Campsite delivery.
3	2001	Deliver material for base construction.
4	2001	Crew of 4 - Base construction and inspection of robotic work; -Stay in Campsite.
5	2002	Deliver construction material.
6	2002	Crew of 4 - Mission similar to Flight 4.
7	2003	Cargo delivery.
8	2003	Crew mission.
9	2004	Delivery of LLOX plant construction material.
10	2004	Crew of 4 - Inspection and construction of LLOX plant and lunar base.
11	2005	Deliver materials for LLOX plant and habitats
12	2005	First crew of 6.
13	2006	Cargo delivery of construction materials.
14	2006	Crew of 6 - Continues aconstruction.

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Lunar Flights - Medium Scale

Flight Number	Year	Mission Description
15-16	2007	Cargo deliveries of materials and supplies.
18	2007	Crew of 6 - 1 year stay.
19-20	2008	Deliver material for base construction.
21	2008	Crew of 6 - Base construction and inspection of robotic work; -Stay at the base.
22-23	2009	Deliver construction material.
24	2009	Crew of 6 - Mission similar to Flight 21.
25	2010	Cargo delivery - Uses LLOX
26	2010	First crew mission to use LLOX.

In the period of **2010 - 2025**, there will be **15** crew missions which use lunar oxygen as the oxidizer. Also during this time, a minimum of **15** LLOX cargo flights have been manifested. This totals to **56** flights during the program. It is possible to have more flights, but these have not been accounted for. If large payloads are necessary, tandem LTVs should be used in place of the LLOX flights.

Crew Rotation - Medium Scale

Launch	Return	Stay	Base Crew
00 00 00 00	00 00 00 00	n/a	0
01 01 01 01	01 01 01 01	n/a	0
02 02 02 02	02 02 02 02	n/a	0
03 03 03 03	03 03 03 03	n/a	0
04 04 04 04	04 04 04 04	n/a	0
05 05 05 05 05 05	05 05 05 05 05 05	n/a	0
06 06 06 06 06 06	06 06 06 06 06 06	n/a	0
07 07 07 07 07 07	07 07 07	07 07 07	3
08 08 08 08 08 08	07 07 07 08 08 08	08 08 08	3
09 09 09 09 09 09	08 08 08	09 09 09 09 09 09	6
10 10 10 10 10 10	09 09 09 09 09 09	10 10 10 10 10 10	6
11 11 11 11 11 11	10 10 10 10 10 10	11 11 11 11 11 11	6
12 12 12 12 12 12	11 11 11 11 11 11	12 12 12 12 12 12	6
13 13 13 13 13 13	12 12 12 12 12 12	13 13 13 13 13 13	6

Note: These numbers indicate the year each mission was launched from Earth beginning in 2000. This numbering scheme remains constant for the return trip and stay times.

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Crew Rotation - Medium Scale

Launch	Return	Stay	Base Crew
14 14 14 14 14 14	13 13 13 13 13 13	14 14 14 14 14 14	6
15 15 15 15 15 15	14 14 14 14 14 14	15 15 15 15 15 15	6
16 16 16 16 16 16	15 15 15 15 15 15	16 16 16 16 16 16	6
17 17 17 17 17 17	16 16 16 16 16 16	17 17 17 17 17 17	6
18 18 18 18 18 18	17 17 17 17 17 17	18 18 18 18 18 18	6
19 19 19 19 19 19	18 18 18 18 18 18	19 19 19 19 19 19	6
20 20 20 20 20 20	19 19 19 19 19 19	20 20 20 20 20 20	6
21 21 21 21 21 21	20 20 20 20 20 20	21 21 21 21 21 21	6
22 22 22 22 22 22	21 21 21 21 21 21	22 22 22 22 22 22	6
23 23 23 23 23 23	22 22 22 22 22 22	23 23 23 23 23 23	6
24 24 24 24 24 24	23 23 23 23 23 23	24 24 24 24 24 24	6
25 25 25 25 25 25	24 24 24 24 24 24	25 25 25 25 25 25	6

Note: These numbers indicate the year each mission was launched from Earth beginning in 2000. This numbering scheme remains constant for the return trip and stay times.

,STCAEM/tls/05Sept90

Lunar Manifest Worksheet - Medium Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2000	Crew Sortie	LTVT Dir/fo	15 t	2 LTVs, 1 Brake, Landing Legs	2 LTVs, 1 Brake, Landing Legs	202 t	255 t	3	4 HLLV 1 Crew	4 Crew, 4 - 6 days on LS
2001	Crew	LTVT Dir/fo	15 t	2 LTVs, 1 Brake, Landing Legs	1LTV, 1 Brake, Landing Legs	220 t	277 t	3	3 HLLV 1 Crew	4 Crew, 40 days on LS
2002	Crew	LOR	3.5 t	1 LTV, 1 LEV, 1 LTVCM, 1 LEVCM, Aerobrake	1 LTV, 1 LEV, 1 LEVCM, 1 LTVCM, Aerobrake	132 t	174 t	2	4 HLLV 1 Crew	4 Crew, 40 days on LS
2003-2004	Crew	LOR	10 t	1 LTV, 1 LEV, 1 LTVCM, 1 LEVCM, Aerobrake	None (reuse everything)	134 t	170 t	2	1 Crew (Cargo in STS flight)	4 Crew, 6 months on LS
2005-2006	Crew	LOR	10 t	1 LTV, 1 LEV, 1 LTVCM, 1 LEVCM, Aerobrake	None (reuse everything)	135 t	171 t	2	1 Crew (Cargo in STS flight)	6 Crew, 6 month stay

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Lunar Manifest Worksheet - Medium Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2007	Crew	LOR	3 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	132 t	173 t	2	3 HLLV 1 Crew	6 Crew, 1 year on LS
2008, 2009	Crew	LOR	~~11~~ t 10	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reused LTV and LEV components)	~~137~~ t 134	~~174~~ t 171	2	1 HLLV 1 Crew	6 Crew, 1 year on LS
2010, 2015, 2020, 2025	Crew	LOR	6 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LTV, LTVCM, LEV, LEVCM, Aerobrake	116 t	159 t	2	3 HLLV 1 Crew	6 Crew, 1 year on LS
2011, 2016, 2021	Crew	LOR	15 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reused LTV and LEV components)	94 t	134 t	1	1 Crew (Cargo in STS flight)	6 Crew, 1 year on LS
2012, 2017, 2022	Crew	LOR	15 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reused LTV and LEV components)	94 t	134 t	1	1 Crew (Cargo in STS flight)	6 Crew, 1 year on LS

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Lunar Manifest Worksheet - Medium Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2013, 2018, 2023	Crew	LOR	15 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reused LTV and LEV components and Brake)	94 t	134 t	1	1 Crew (Cargo in STS flight)	6 Crew, 1 year stay
2000	Campsite Delivery	LTVT Dir/fo	24 t + Camp-site	2 LTVs, 1 Brake, Landing Legs	2 LTVs, Landing Legs, Brake, Campsite	206 t	290 t	3	4 HLLV	N/A
2001-2009	Cargo	LTVT Dir/fo	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs (reuse 1 LTV and Brake)	206 t	290 t	2	2 HLLV	N/A
2007-2009	Cargo	LTVT Dir/fo	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs (reuse 1 LTV and Brake)	206 t	290 t	2	2 HLLV	N/A
2010	Cargo	LOR	15 t	1 LTV, 1 Brake, 1 LTVCM	1 LTV, 1 LEV, Aerobrake	99 t	135 t	1	3 HLLV	N/A
2011	Cargo	LOR	25 t	1 LTV, 1 Brake, 1 LTVCM	None (reuse everything)	81 t	120 t	1	1 HLLV	N/A

/STCAEM/tls/28Aug90

Lunar Manifest Worksheet - Medium Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population

After the initial 700 tonnes of material have been delivered to the lunar surface, cargo flights may use Lunar LOX as the oxidizer. If large payloads are necessary, the direct, tandem LTV flights can be used. Otherwise, the cargo missions will be similar to those of 2010 and 2011, depending on the LTV and LEV requirements.

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Lunar Flights - Large Scale

ASSUMPTIONS:

- Flights begin in 2000.
- Minimum of one crew flight per year.
- Rate of cargo flights vary
- Beginning 2015, 80% of habitat facilities are derived from in-situ resources.
- Logistics:
  5 kg/person-day and a 2 year stay by 2010
  3 kg/person-day and a 3 year stay by 2015
  2 kg/person-day and a 4 year stay by 2020
- For industrialization, assume
  Production plant mass = 2 * (annual production)
  Lunar oxygen plant mass = 1.2 * (annual production)
- Additional crew flights are not required to remain for long durations on the lunar surface. These missions may be "visitations" for scientific purposes.

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Mass Required On the Lunar Surface

By 2010	(both Medium and Large Scale Programs)	
	Life Support Hardware	31 t
	LLOX Plant Materials	250 t
	Habitat Structures	250 t
	Additional Mass	169 t
	Total	**700 t**
By 2015	(Large Scale Program only)	
	Life Support Hardware	100 t
	LLOX Plant	250 t
	Habitat Structures	750 t
	Helium-3 Plant Materials	500 t
	Additional Mass	400 t
	Total	**2000 t**
By 2025	(Large Scale Program only)	
	Life Support Hardware	250 t
	LLOX Plant	300 t
	Habitat Structures	1000 t
	Helium-3 Plant Materials	5000 t
	Additional Mass	950 t
	Total	**7500 t**

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Cargo Delivery Requirements - Large Scale

By 2010

- 700 tonnes of material on the lunar surface.
- Use tandem LTVs capable of 55 ton deliveries.
- Requires 12 cargo flights and 1 Campsite delivery.

By 2015

- Additional 2300 tonnes of material are needed
- In-situ resource utilization
  - -- 40% of the additonal 500 tonnes for the habitat modules
  - -- Assume 20% of the Helium-3 plant mass.
- Must deliver 1000 tonnes from the Earth.
- Requires 19 tandem LTV cargo flights.

By 2025

- 4500 tonnes of addtional material are needed.
- In-situ resource utilization
  - -- 50% of the new life support system hardware
  - -- 80% of the remaining habitat mass
  - -- 50% of the additional Helium-3 plant mass
- Must deliver 2975 tonnes from the Earth.
- Requires 54 tandem LTV cargo flights
- May use additional LLOX cargo flights (these have not been manifested)

Lunar Flights - Large Scale

Flight Number	Year	Mission Description
1	2000	Crew Sortie
2	2000	Campsite delivery.
3	2001	Deliver material for base construction.
4	2001	Crew of 4 - Base construction and inspection of robotic work
5	2002	Crew of 4 - Base construction and inspection of robotic work; -Stay in Campsite.
6	2002	Cargo delivery for LLOX plant construction and base build-up.
7	2003	Delivery of LLOX plant and base construction material.
8	2003	Crew of 4 - Inspection and construction of LLOX plant.
9	2004	Deliver materials for production plant, LLOX plant, and base.
10	2004	Crew of 4 - First to stay at the base (stay time of 6 months) - Continues construction and base build-up.
11	2005	Cargo delivery of construction materials and crew supplies
12	2005	Crew of 6 - 3 will remain 6 months while 3 will remain for 1 year.
13	2006	Deliver material for base construction.
14	2006	Crew of 6 - All 6 will remain on surface; - Return crew from previous flight.

Lunar Flights - Large Scale

Flight Number	Year	Mission Description
15	2007	Deliver construction materials and supplies.
16	2007	Crew of 6 - Replaces crew at the base for 1 year.
17	2007	Delivery of materials and supplies.
18-19	2008	Deliver materials for construction.
20	2008	Crew of 6 - Change-out of base personnel
21-22	2009	Cargo delivery.
23	2009	Crew of 6 - Replaces base personnel; - Stay time of 1 year.
24-27	2010	Cargo delivery.
28	2010	Crew of 6 - Change-out of base crew.
29-32	2011	Deliver material for base expansion, production plant
33	2011	Crew of 6 - Crew change-out at the base; - Only 3 from previous crew returns to Earth. - Base now houses 9 crew.
34-37	2012	Deliver material for base expansion.
38	2012	Crew of 6 - Replace some of base personnel; - 9 remain on surface
39-42	2013	Cargo delivery
43	2013	Crew of 6

Lunar Flights - Large Scale

Flight Number	Year	Mission Description
44-46	2014	Cargo delivery - plant construction and base build-up.
47	2014	Crew of 6
48-52	2015	Cargo delivery
53	2015	Crew of 6
54	2016	Crew of 6 - Change-out of base personnel
55-59	2016	Cargo delivery.
60-65	2017	Cargo delivery.
66	2017	Crew of 6 - Replaces base personnel;
67-72	2018	Cargo delivery.
73	2018	Crew of 6 - Change-out of base crew.
74-79	2019	Deliver material for base expansion.
80	2019	Crew of 6 - Replace some of base personnel;
81-86	2020	Cargo delivery
87	2020	Crew of 6
88-92	2021	Cargo delivery - plant construction and base build-up.
93	2021	Crew of 6

Lunar Flights - Large Scale

Flight Number	Year	Mission Description
94-98	2022	Cargo delivery
99	2022	Crew of 6
100	2023	Crew of 6 - Change-out of base personnel
101-105	2023	Cargo delivery.
106-110	2024	Cargo delivery.
111	2024	Crew of 6 - Change-out of base personnel.
112	2025	Crew of 6 - Replaces base personnel.
113	2025	Cargo delivery.

Crew Rotation - Large Scale

Launch	Return	Stay	Base Crew
00 00 00 00	00 00 00 00	n/a	0
01 01 01 01	01 01 01 01	n/a	0
02 02 02 02	02 02 02 02	n/a	0
03 03 03 03	03 03 03 03	n/a	0
04 04 04 04	04 04 04 04	n/a	0
05 05 05 05 05 05	05 05 05 05 05 05	n/a	0
06 06 06 06 06 06	06 06 06 06 06 06	n/a	0
07 07 07 07 07 07	07 07 07	07 07 07	3
08 08 08 08 08 08	07 07 07 08 08 08	08 08 08	3
09 09 09 09 09 09	08 08 08	09 09 09 09 09 09	6
10 10 10 10 10 10	09 09 09 09 09 09	10 10 10 10 10 10	6
11 11 11 11 11 11	10 10 10	10 10 10 11 11 11 11 11 11	9
12 12 12 12 12 12	10 10 10	11 11 11 11 11 11 12 12 12 12 12 12	12
13 13 13 13 13 13	11 11 11 11 11 11	12 12 12 12 12 12 13 13 13 13 13 13	12

Note: These numbers indicate the year each mission was launched from Earth beginning in 2000. This numbering scheme remains constant for the return trip and stay times.

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Crew Rotation - Large Scale

Launch	Return	Stay	Base Crew
14 14 14 14 14 14	12 12 12 12 12 12	13 13 13 13 13 13 14 14 14 14 14 14	12
15 15 15 15 15 15	13 13 13 13 13 13	14 14 14 14 14 14 15 15 15 15 15 15	12
16 16 16 16 16 16	14 14 14 14 14 14	15 15 15 15 15 15 16 16 16 16 16 16	12
17 17 17 17 17 17	15 15 15	15 15 15 16 16 16 16 16 16 17 17 17 17 17 17	15
18 18 18 18 18 18	15 15 15	16 16 16 16 16 16 17 17 17 17 17 17 18 18 18 18 18 18	18
19 19 19 19 19 19	16 16 16 16 16 16	17 17 17 17 17 17 18 18 18 18 18 18 19 19 19 19 19 19	18

Note: These numbers indicate the year each mission was launched from Earth beginning in 2000. This numbering scheme remains constant for the return trip and stay times.

Lunar Manifest Worksheet - Large Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2000	Sortie	LTVT Dir/fo	10 t	2 LTVs, 1 Brake, Landing Legs, Excursion Cab, ACRV	2 LTVs, 1 Brake, Landing Legs, ACRV, Excursion Cab	209 t	261 t	3	3 HLLV 1 Crew	4 Crew, 4 - 6 days on LS
2001	Crew	LTVT Dir/fo	10 t	2 LTVs, 1 Brake, Landing Legs, Excursion Cab, ACRV	1 LTV, Excursion Cab, ACRV, Landing Legs (reuse 1 LTV and Brake)	209 t	261 t	3	2 HLLV 1 Crew	4 Crew, 40 days on LS
2002	Crew	LOR	3 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LEV, LTVCM, LEVCM, Brake (reuse LTV)	131 t	171 t	2	2 HLLV 1 Crew	4 Crew, 40 days on LS
2003	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reuse everything)	134 t	170 t	2	1 Crew (Cargo in STS flight)	4 Crew, 40 days on LS
2004	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LTV, LTVCM (reuse LEV components and Brake)	131 t	171 t	2	1 Crew (Cargo in STS flight)	4 Crew, 40 days on LS

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Lunar Manifest Worksheet - Large Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2005	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reuse everything)	135 t	171 t	2	1 Crew (Cargo in STS flight)	6 Crew, 6 months on LS
2006	Crew	LOR	3 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LEV, LEVCM, Brake (reuse LTV components)	132 t	173 t	2	2 HLLV 1 Crew	6 Crew, 6 months on LS
2007	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reuse everything)	135 t	171 t	2	1 Crew (Cargo in STS flight)	6 Crew, 3 remain at lunar base
2008	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reuse everything and refurbish brake)	135 t	171 t	2	1 Crew (Cargo in STS flight)	6 Crew, 3 remain at lunar base
2009	Crew	LOR	10 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LTV, LTVCM (reuse LEV components and brake)	135 t	171 t	2	1 HLLV 1 Crew (Cargo in STS flight)	6 Crew, 6 remain at lunar base

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Lunar Manifest Worksheet - Large Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2010	Crew (LLOX mission)	LOR	15 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	None (reuse everything)	93 t	134 t	1	1 Crew (Cargo in STS flight)	6 Crew, 6 remain at lunar base
2011	Crew (LLOX mission)	LOR	8 t	1 LTV, 1 Brake, 1 LTVCM, 1 LEV, 1 LEVCM	LEV, LEVCM, Brake (reuse LTV components)	111 t	167 t	2	1 HLLV 1 Crew (Cargo in STS flight)	6 Crew, 9 remain at lunar base
2012-2015	Crew	LOR	15 t							
2016	Crew	LOR	8 t	Transportation requirements are the same for the remainder of the program (for LOR type missions).	These missions will be very similiar to the 2010 and 2011 missions, which is apparent by the cargo masses. The missions in 2014, 2019, and 2024 will require a new LTV and Aerobrake. This increases the number of ETO launches by 2 for those missions. The propellant loadings are not affected by this. A crew of 6 launches from the Earth in each mission. However, not al: will return immediately. By 2025, the base will have a population of 18 members.					
2017-2020	Crew	LOR	15 t							
2021	Crew	LOR	8 t							
2022-2025	Crew	LOR	15 t							

Lunar Manifest Worksheet - Large Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2000	Campsite Delivery	LTVT Dir/exp	24 t + Camp-site	2 LTVs, 1 Brake, Landing Legs	2 LTVs, 1 Brake, Landing Legs	206 t	290 t	3	4 HLLV	N/A
2001, 2006, 2009-2010, 2012-2025	Cargo	LTVT Dir/exp	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs, (reuse Brake and 1 LTV)	206 t	290 t	3	2 HLLV	N/A
2002, 2007, 2009, 2011-2013, 2015-2017, 2017-2024	Cargo	LTVT Dir/exp	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs, (reuse Brake and 1 LTV)	206 t	290 t	3	2 HLLV	N/A
2003, 2007, 2010-2013, 2015-2018, 2018-2024	Cargo	LTVT Dir/exp	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs, (reuse Brake and 1 LTV)	206 t	290 t	3	2 HLLV	N/A

Lunar Manifest Worksheet - Large Scale

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2004, 2008, 2010-2012, 2014-2019, 2019-2024	Cargo	LTVT Dir/exp	55 t	2 LTVs, 1 Brake, Landing Legs	1 LTV, Landing Legs, (reuse Brake and 1 LTV)	206 t	290 t	3	2 HLLV	N/A
2005, 2008, 2010, 2011, 2013-2020, 2020-2024	Cargo	LTVT Dir/exp	55 t	2 LTVs, 1 Brake, Landing Legs	2 LTVs, Landing Legs, 1 Brake	206 t	290 t	3	4 HLLV	N/A

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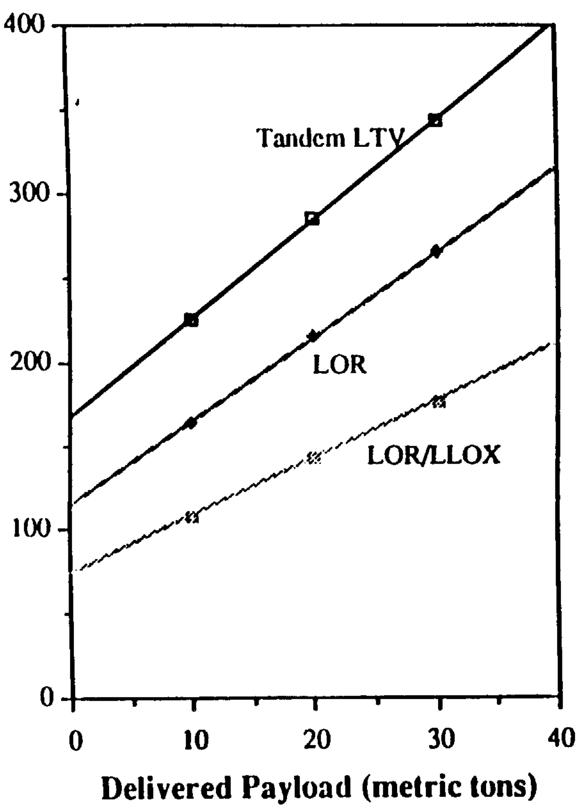
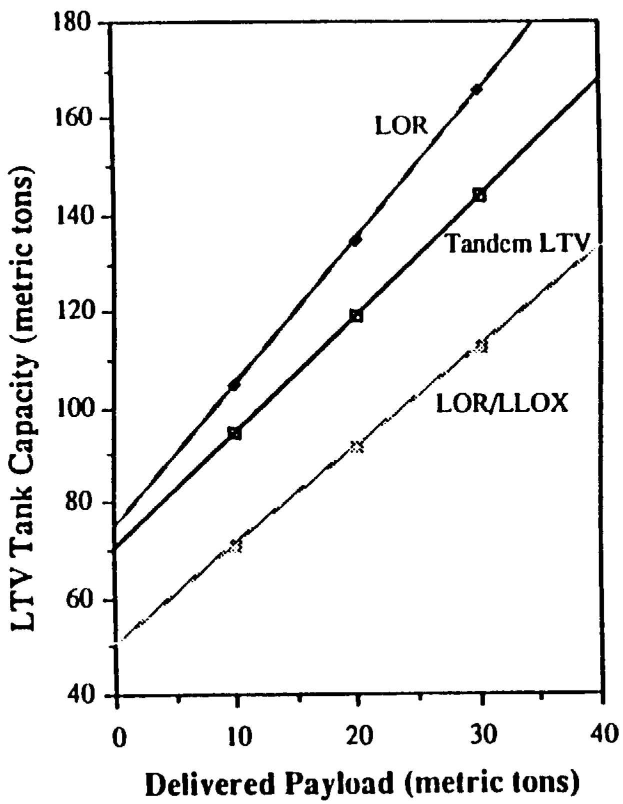
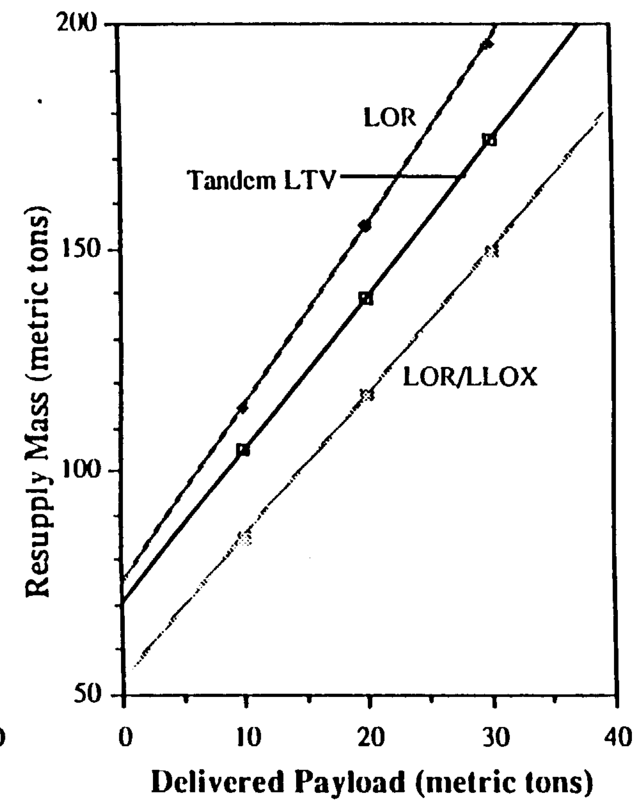
Lunar Modes Performance

Crew Mission, 1 Ton Returned

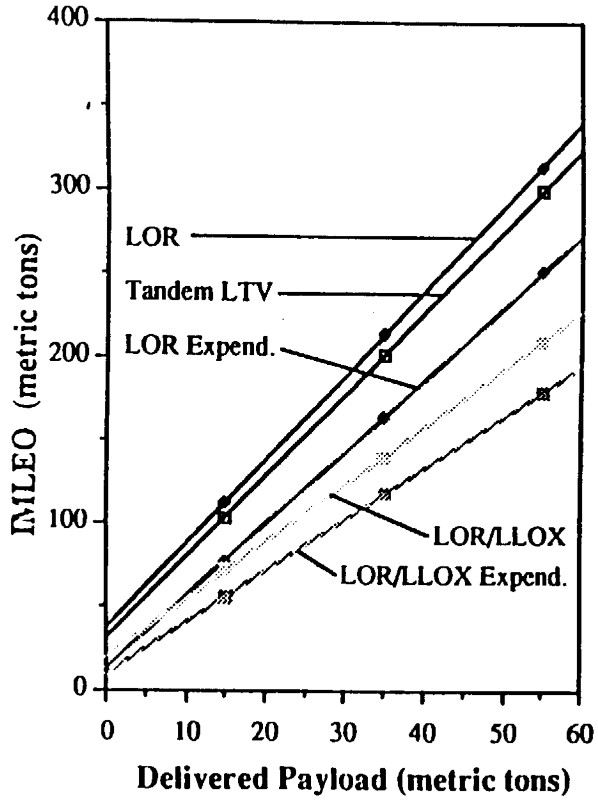


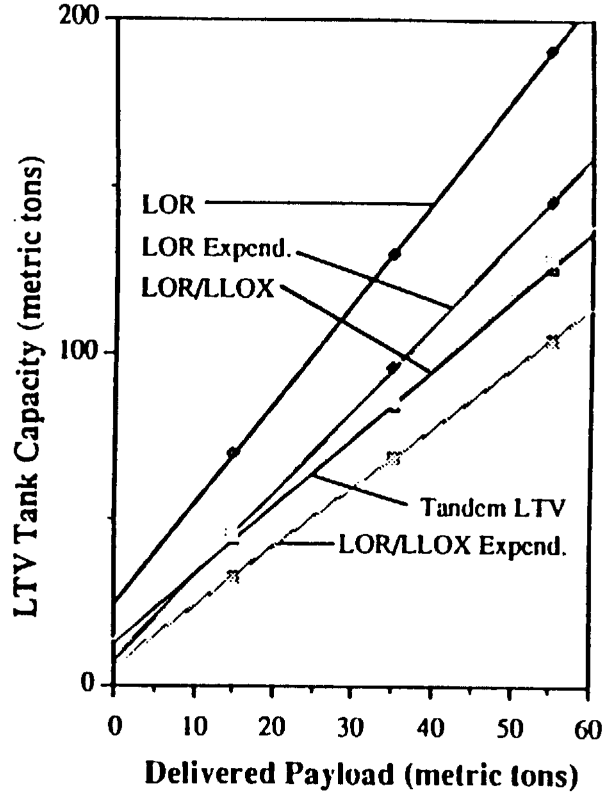


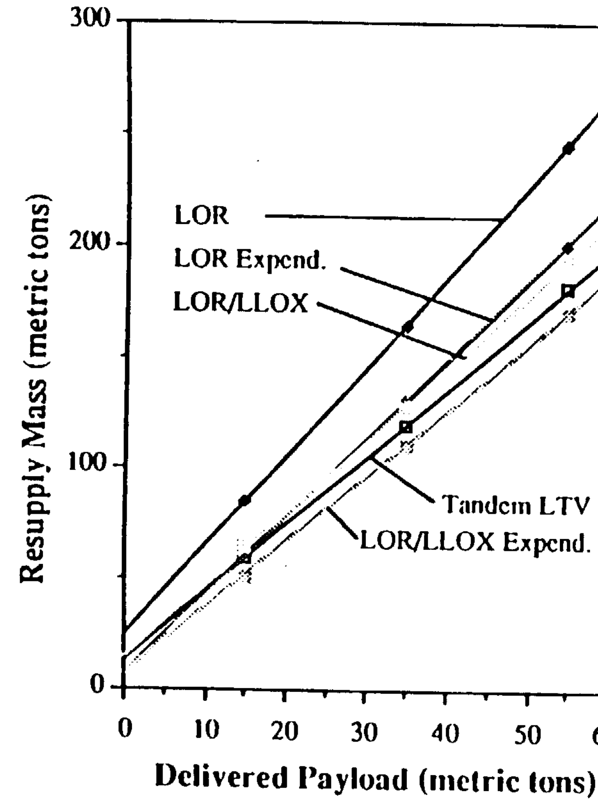


Notes:
- Crew size of 6
- LEV crew module of 4.847 tons
- 1 ton of additional consumables
- Isp of 475 seconds
- LTV crew module of 8.911 tons

Lunar Modes Performance
Cargo Delivery Mission







Lunar Manifest Worksheet

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2004	1st Manned	LTVT Dir/fo	5 t	2-LTV, 1-Brake, 1-mod. ACRV, Service Module	2-LTV, 1-Brake, 1-mod. ACRV, Service Module, LTV Landing Legs	185 t	230 t	3	4 HLLV 1 Crew	4 Crew, 4 - 6 days on LS
2005	Campsite Delivery	LTVT Dir,exp	15 t + Camp-site	2 LTV 1-Brake	1-LTV, Campsite, LTV Landing Legs, 2 unpressurized rovers	194 t	270 t	3	3 HLLV	N/A
2006	1st Overnite Stay	LTVT Dir/fo	10 t	2-LTV,1-Brake, 1-mod. ACRV, Service Module	Reused LTVs and Brake, 1-mod. ACRV, Landing Legs, Service Module, Spare LTV and Brake	208 t	261 t	3	3 HLLV 1 Crew	4 Crew, 40 days on LS
2008	Science Cargo	LTVT Dir,exp	45 t	2-LTV 1-Brake	Science Cargo, 1-LTV, 2 pressurized rovers, Replacable Camp-site power supply	191 t	266 t	3	2 HLLV	N/A
2009	Crew	LTVT Dir/fo	10 t	2-LTV, 1-Brake, 1-mod. ACRV, Service Module	1-LTV, 1-mod. ACRV, Landing Legs, Service Module, replacable power supply	208 t	261 t	3	2 HLLV	4 Crew, 40 days on LS

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Lunar Manifest Worksheet

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2004	1st Manned	LOR	5 t	1-LTV, 1-Brake, 1-LTVCM, 1-LEV, 1-LEVCM	1-LTV, 1-Brake, 1-LTVCM, 1-LEV, 1-LEVCM	136 t	180 t	2	3 HLLV 1 Crew	4 Crew, 4 - 6 days on LS
2005	Campsite Delivery	LOR	10 t + Camp-site	1 LTV, 1-Brake, 1-LEV	Campsite, 2 unpressurized rovers, Spare LTV and Brake (reused LTV and Brake)	209 t	276 t	3	4 HLLV	N/A
2006	1st Overnite Stay	LOR	10 t	1-LTV, 1-Brake, 1-LTVCM, 1-LEV, 1-LEVCM	None (reused LTV and LEV components)	142 t	183 t	2	1 HLLV 1 Crew	4 Crew, 40 days on LS
2008	Science Cargo	LOR	44 t	1 LTV, 1-Brake, 1-LEV	Science Cargo, 2 pressurized rovers, Replacable Camp-site power supply, (reused LTV and LEV)	218 t	288 t	3	2 HLLV	N/A
2009	Crew	LOR	10 t	1-LTV, 1-Brake, 1-LTVCM, 1-LEV, 1-LEVCM	Replacable power supply, Refurbish Brake (Reused LTV and LEV)	142 t	183 t	2	1 HLLV 1 Crew	4 Crew, 40 days on LS

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Lunar Manifest Worksheet

Year	Mission	Mode	Cargo Mass	Transportation Elements	Hardware Consumed (ETO Launch Rqts)	Propellant to LEO	Total to LEO	Tankers in LEO	ETO Launches	Lunar Population
2004	1st Manned	LOR	5 t	1-LTV, 1-Brake, 1-LTVCM, 1-LEV, 1-LEVCM	1-LTV, 1-Brake, 1-LTVCM, 1-LEV, 1-LEVCM	136 t	180 t	2	3 HLLV 1 Crew	4 Crew, 4 - 6 days on LS
2005	Campsite Delivery	LOR/ exp	20 t + Camp-site	1 LTV, 1-Brake, 1-LEV	Campsite, 2 unpressurized rovers (reuse LTV and Brake)	185 t	252 t	3	2 HLLV	N/A
2006	1st Overnite Stay	LOR	10 t	1-LTV, 1-Brake, 1-LTVCM, 1-LEV, 1-LEVCM	1-LTV, 1-Brake, 1-LTVCM, 1-LEV, 1-LEVCM	142 t	183 t	2	3 HLLV 1 Crew	4 Crew, 40 days on LS
2008	Science Cargo	LOR/ exp	50 t	1 LTV, 1-Brake, 1-LEV	Science Cargo, 2 pressurized rovers, Replacable Camp-site power supply, (reused LTV and LEV)	182t	248 t	3	2 HLLV	N/A
2009	Crew	LOR	10 t	1-LTV, 1-Brake, 1-LTVCM, 1-LEV, 1-LEVCM	1-LTV, 1-Brake, 1-LTVCM, 1-LEV, 1-LEVCM, replacable power supply	142 t	183 t	2	3 HLLV 1 Crew	4 Crew, 40 days on LS

Lunar Manifest Worksheet

YEAR	2004	2005	2006	2008	2009
MISSION	1st Manned	Campsite Delivery	1st Overnite Stay	Science Cargo	Crew
MODE	LTVT Dir/FO	LTVT Dir/FO	LTVT Dir/FO	LTVT Dir/FO	LTVT Dir/FO
FLIGHT HLLV 1	LTV	LTV	LTV (spare)	LTV	LTV
2	LTV	Campsite	ACRV, S/M, Payload	Cargo, Rovers	ACRV, S/M, Payload
3	ACRV, S/M, Payload	Payload	Aerobrake		
4	Aerobrake				
CREW 1	Crew of 4	N/A	Crew of 4	N/A	Crew of 4
TANKERS	3	3	3	3	2

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Lunar Manifest Worksheet

YEAR	2004	2005	2006	2008	2009
MISSION	1st Manned	Campsite Delivery	1st Overnite Stay	Science Cargo	Crew
MODE	LOR	LOR	LOR	LOR	LOR
FLIGHT HLLV 1	LTV	LTV (Spare)	Payload	Science Equipment	Cargo
2	LEV	Campsite		Payload	
3	Aerobrake	Payload			
4		Aerobrake			
CREW 1	Crew of 4	N/A	Crew of 4	N/A	Crew of 4
TANKERS	2	3	2	3	2

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Lunar Manifest Worksheet

YEAR	2004	2005	2006	2008	2009
MISSION	1st Manned	Campsite Delivery	1st Overnite Stay	Science Cargo	Crew
MODE	LOR	LOR/ Expendable	LOR	LOR/ Expendable	LOR
FLIGHT HLLV 1	LTV	Payload	LTV	LTV	LTV
2	LEV Payload	Campsite	LEV Payload	Payload	LEV Payload
3	Aerobrake		Aerobrake		Aerobrake
4					
CREW 1	Crew of 4	N/A	Crew of 4	N/A	Crew of 4
TANKERS	2	3	2	3	2

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END DATE MAY 11, 1993